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USER DOCUMENTATION OF THE CTA PROGRAM

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FOR THE COMMANDER

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13. ABSTRACT *(Maximum 200 words)*

This report is a user's manual for the CTA program.

CTA is a FORTRAN program which converts Cartesian aero-angle aerodynamic coefficients, as generated by the Missile DATCOM aerodynamic prediction code, to polar aeroballistic angle coefficients. The output format of the aeroballistic coefficients matches the format used in the CADAC 6 DOF flyout simulation.

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LIST OF SYMBOLS AND ABBREVIATIONS

1^B, 2^B, 3^B - Body axis
1^S, 2^S, 3^S - Stability axis
AOA - Angle of Attack
CA - Cartesian Aero-Angles
CADAC - Computer Aided Design of Armament Concepts simulation
CG - Center of Gravity
DOF - Degrees Of Freedom
MD - Missile DATCOM
MRC - Moment Reference Center
M# - Mach number
NDS - Number of datasets
NFIN - Number of finset to be deflected
PA - Polar Aeroballistic Angles
$[V_B{}^A]$ - velocity vector of the relative wind
Y_{cent} - distance from body centerline to centroid of control fin - l_{ref} units
ft - feet
in - inches
l_{ref} - reference length
δp - roll control surface deflection
δq - pitch control surface deflection
δr - yaw control surface deflection
α - angle of attack, degrees
α' - total aerodynamic angle of attack, degrees
β - side-slip angle, degrees
ϕ' - total aerodynamic roll angle, degrees

1. INTRODUCTION

The aerodynamic forces and moments on a missile are produced by its relative motion with respect to the air and depend on the orientation of the missile with respect to the airflow. In uniform airflow these forces and moments are unchanged after a rotation around the free-stream velocity vector. Therefore, two orientation angles, with respect to the relative wind, $[V_B{}^A]$, are needed to specify the aerodynamic forces and moments.[1] There are two methods of defining the two orientation angles. In aircraft and missile aerodynamics, the Cartesian aero-angles (CA) are most commonly used. An alternate used in missile aerodynamics is the polar aeroballistic angles (PA). Converting CA orientation angles to PA and vice-versa is simple but since the equations of motion and thus force and moment coefficients associated with each are different, the complete conversion process is more complex.

The most frequently used method of predicting the aerodynamic coefficients for missiles is the Missile DATCOM (MD) semi-empirical aerodynamic prediction code. In MD the body orientation can be input in either of the two methods discussed above, but the aerodynamic coefficients output are those associated with CA.

The CA system works well with the equations of motion for 3 degree of freedom (DOF) flyout simulations. For a six DOF simulation of a tetragonally symmetrical missile, the equations of motion are simplified if the PA system is used. A program which converts MD output from CA coefficients to PA coefficients would streamline the work required to run such a simulation when wind tunnel aerodynamic data are not available.

2. REFERENCE SYSTEMS & EQUATIONS

In converting from one reference system to another, the basis for each system must be defined. Sections 2.1 through 2.3 review the reference systems, force and moment coefficients as well as the equations of motion for the PA system.

2.1 BODY-AXIS SYSTEM

In the body-axis system shown in Figure 1, the axes are fixed to the body with the origin at the center of mass of the vehicle with the body x-axis (1^B) pointing forward and the body z-axis (3^B) pointing down. These axes usually coincide with the principal axes of inertia.

2.2 CA SYSTEM

The CA orientation angles are also shown in Figure 1. The orientation of the vehicle relative to the resultant wind, $[V_B{}^A]$, is defined by the angle of attack, α, and the sideslip angle, β. The first rotation defines the stability axis, 1^S, where α is the angle between the body-fixed x-axis and the stability x-axis. Alpha is positive if the rotation about the body-fixed y-axis (2^B) is negative, thus a positive α is shown. The second rotation leads to the wind axes, where the sideslip angle, β, is the angle between the stability x-axis (1^S) and the wind axis. Beta is positive if the rotation about the stability z-axis (3^S) is positive, thus a positive β is shown.

The force and moment coefficients for the CA system along with their MD variable names are listed in the Table 1. All but C_{lp} is calculated by MD.

Table 1. CA Coefficients and MD Variable Names.

Force Coefficients:		Definition:
C_A	CA	Axial
C_N	CN	Normal
C_D	CD	Drag
C_L	CL	Lift
C_Y	CY	Side
X_{cp}	X-C.P.	Center of Pressure (Cal. from MRC)
Moment Coefficients:		
C_l	CLL	Roll
C_m	CM	Pitch
C_n	CLN	Yaw
Derivatives:		
$C_{N\alpha}$	CNA	Normal Force coef. deriv. wrt Alpha
$C_{Y\beta}$	CYB	Side Force coef. deriv. wrt Beta
$C_{m\alpha}$	CMA	Pitching Moment coef. deriv. wrt Alpha
$C_{n\beta}$	CLNB	Yawing Moment coef. deriv. wrt Beta
$C_{l\beta}$	CLLB	Rolling Moment coef. deriv. wrt Beta
Dynamic Derivatives:		
C_{lp}		Roll Moment coef. deriv wrt to Roll Rate
C_{Nq}	CNQ	Normal Force coef. deriv due to Pitch Rate
$C_{N\dot{\alpha}}$	CNAD	Normal Force coef. deriv wrt to accel in AOA
C_{mq}	CMQ	Pitching Moment coef. deriv. wrt Pitch Rate
$C_{m\dot{\alpha}}$	CMAD	Pitching Moment coef. deriv. wrt accel. in AOA

2.3 PA SYSTEM

The PA system is also a body-axis system but the orientation of the vehicle relative to the resultant wind, $[V_B{}^A]$, is defined by the total angle of attack, α' and aerodynamic roll angle, ϕ', as shown in Figure 2. The total angle of attack is defined by the angle between the longitudinal axis of missile and resultant wind. The total angle of attack is always positive. The aerodynamic roll angle is defined as the angle between the reference plane and plane of resultant wind and is considered positive if the rotation about the body-fixed x-axis is positive (clockwise when looking in the +X direction), thus a positive ϕ' is shown.

The following equations convert between the CA and PA orientation angles.

$$\alpha = \tan^{-1} (\tan \alpha' \cos \phi')$$

$$\beta = \sin^{-1} (\sin \alpha' \sin \phi')$$

$$\alpha' = \cos^{-1} (\cos \alpha \cos \beta)$$

$$\phi' = \tan^{-1} (\tan \beta / \sin \alpha)$$

The PA 6 DOF aerodynamic equations for a missile with tetragonal symmetry are as follows.[4]

Force Coefficients

$$C_A = C_{Ao}(M) + C_{A_{\alpha'}}(M)\alpha' + C_{A\delta^2_{eff}}(M)\delta^2_{eff} \quad ; \quad \delta_{eff} = \frac{|\delta q| + |\delta r|}{2}$$

$$C'_Y = \Delta C'_{Y,\phi'}(M,\alpha')\sin 4\phi' + C'_{Y_{\delta r}}(M)\delta r$$

$$C'_N = C'_{No}(M,\alpha') + \Delta C'_{N,\phi'}(M,\alpha')\sin^2 2\phi' + C'_{N_{\delta q}}(M)\delta q$$

Moment Coefficients

$$C_l = C_{l,\phi'_{\alpha 2}}(M)\alpha'^2 \sin 4\phi' + C_{l_p}(M)\frac{pl}{2V} + C_{l_{\delta p}}(M)\delta p$$

$$C'_m = C'_m(M,\alpha') + \Delta C'_{m,\phi'}(M,\alpha')\sin^2 2\phi' + C'_{m_q}(M)\frac{ql}{2V} + C'_{m_{\delta q}}(M)\delta q$$

$$C'_n = \Delta C'_{n,\phi'}(M,\alpha')\sin 4\phi' + C'_{n_r}(M)\frac{rl}{2V} + C'_{n_{\delta r}}(M)\delta r$$

The force and moment coefficients used in the PA equations above are listed in Table 2 below, along with the variable name used in the CTA code.

2.3 CONTROL SURFACE DEFLECTIONS

The MD runs used to calculate the coefficients in Table 2 require roll, pitch and yaw control surface deflections. Care must be taken to insure that the proper deflections are specified because fin numbering and definition of a positive deflection varies. In MD, a positive deflection angle produces a negative body axis rolling moment at zero angle of attack. The roll, pitch, and yaw control for the MD convention can be defined in terms of individual surface deflections by the formulas for δp, δq, δr given below and illustrated in Figure 3.

$$\delta_p = \frac{-\delta_1 - \delta_2 - \delta_3 - \delta_4}{4}$$

$$\delta_q = \frac{-\delta_1 - \delta_2 + \delta_3 + \delta_4}{4}$$

$$\delta_r = \frac{-\delta_1 + \delta_2 + \delta_3 - \delta_4}{4}$$

Table 2. PA Coefficients and CTA Variable Names.

Axial Force Coefficient:		Definition:
C_{Ao}	CA	Axial Force coef. at zero α' - f{M#}
$C_{A_{\alpha'}}$	CAA	Variation of C_A with α' - f{M#}
$C_{A_{\delta^2 eff}}$	CAD	Variation of C_A with Effective Control Surface Deflection - f{M#}
Side Force Coefficient		
$\Delta C'_{Y,\phi'}$	CYP	Side Force coef. at $\phi' = 22.5^{\circ}$ - f{M#, α'}
$C'_{Y_{\delta r}}$	= CNDQ	Variation of C'_Y with Yaw Control - f{M#}
Normal Force Coefficients:		
C'_{No}	CN	Normal Force coef. at α' - f{M#, α'}
$\Delta C'_{N,\phi'}$	CNP	Variation of C'_N with roll angle (C'_N at ϕ'=45°-C'_N at ϕ'=0°) - f{M#, α'}
$C'_{N_{\delta q}}$	CNDQ	Pitch Control Effectiveness - f{M#}
Rolling Moment Coefficients:		
$C_{l,\phi'_{\alpha^2}}$	CLLAP	Induced Roll Moment - f{M#}
C_{l_p}	CLLP	Roll Damping - f{M#}
$C_{l_{\delta p}}$	CLLDP	Roll Control Effectiveness - f{M#}
Pitching Moment Coefficients:		
C_m	CLM	Pitching Moment coef., wind axis - f{M#, α'}
$\Delta C'_{m,\phi'}$	CLMP	Variaion of C'_m with roll angle (C_m at ϕ'=45° - C_m at ϕ'=0°) - f{M#, α'}
C'_{m_q}	CLMQ	Pitch Damping - f{M#}
$C'_{m_{\delta q}}$	CLMDQ	Pitch Control Effectiveness - f{M#}
Yawing Moment Coefficients:		
$\Delta C'_{n,\phi'}$	CLNP	Yawing Moment coef. at ϕ'=22.5° - f{M#, α'}
C'_{n_r}	= CLMQ	Yaw Damping - f{M#}
$C'_{n_{\delta r}}$	= CLMDQ	Yaw Control Effectiveness - f{M#,}

3. CTA PROGRAM

The CTA program calculates the PA coefficients listed in Table 2 from a standard MD output file, FOR006.DAT. The program carries out following steps:

- reads in data from MD output file (FOR006.DAT) and the CTA.DAT file
- outputs MD data to file (CART.OUT) to check that the data has been properly read
- calculates PA coefficients from CA coefficients
- writes PA coefficients to output file (AEROB.OUT)

A detailed explanation and instructions are in the following sections.

3.1 MD

The MD semi-empirical aerodynamic prediction code estimates the aerodynamics of a wide variety of missile configurations to an accuracy suitable for preliminary missile design[5]. Revision 6/93 is used in this project.

Before running MD, an input file (FOR005.DAT) must be created. It includes geometry and flight conditions. The FOR005.DAT file used in this study is located in Appendix A with a partial listing of the resulting output file (FOR006.DAT) in Appendix B. The general information required to run MD is covered in Reference 5, so only the specific requirements for CTA will be covered here.

The angle of attack sweep and the Mach number for each dataset are in the FLTCON section of FOR005.DAT. Looking at Appendix A, note that both the NMACH and NALPHA inputs are two digits (8 is input as 08). This two digit format must be retained for these two values to be properly read by CTA (no other variables have this requirement). There are two ways to define the orientation angles within MD: ALPHA and BETA or ALPHA and PHI. If PHI is input and non-zero, it is assumed that ALPHA is the total angle of attack (α') and PHI is the aerodynamic roll angle (ϕ'). The CTA code requires the second method.

The baseline case is a roll angle (ϕ') of zero (fins are located 45° from the Y-Z axis at $\phi' = 0$). The next two cases set ϕ' to 22.5° and 45° respectively. Next, ϕ' is returned to zero and the four fin deflection cases are run, with δ_p, δ_q, δ_r and δ_{eff} set to 5° respectively. For the last case, δ_p is set to 5° and ϕ' to 22.5°. This order must be retained. The total number of data sets (NDS), 64 in this case, will change only if the number of Mach numbers change (NDS = 8 * NMACH). Table 3 below illustrates the required sequence.

Table 3. MD Run Sequence.

DATASET	ϕ'	δ
1 → 8 (8 M#)	0	0
9 → 16	22.5	0
17 → 24	45	0
25 → 32	0	$\delta p = 5°$
33 → 40	0	$\delta q = 5°$
41 → 48	0	$\delta r = 5°$
49 → 56	0	$\delta eff = 5°$
57 → 64	22.5	$\delta p = 5°$

3.2 CTA

The next step is to edit the CTA.DAT file (Appendix C). It contains the total number of datasets (NDS), the distance from the centerline to centroid of control fin (Y_{cent}), and the number of the fin set that is being deflected (NFS). For this case, NDS is 64, Y_{cent} is 0.3751 ft (must be same units as l_{ref}) and NFS is 2.

Finally, the CTA.FOR program is compiled, linked and executed. The two input files required when running CTA.FOR are CTA.DAT and FOR006.DAT previously discussed. When CTA has executed properly, two output files will be created, CART.OUT and AEROB.OUT. CART.OUT (Appendix D) contains the data read in from the MD output file in a headerless format to verify input from FOR006.DAT. AEROB.OUT (Appendix E) contains the PA coefficients in a format similar to that required by the CADAC flyout simulation. The changes that have to be made are

adding a slash at the end of each dataset and shortening some lines to less than 72 columns. The definitions of the variable names used are listed in Table 2.

All data generated by MD and CTA should be reviewed for reasonableness. If the AEROB.DAT data looks questionable, the first thing that should be checked is a match between FOR006.DAT and CART.OUT. If differences are found, CTA has not read the input data properly. This program is not robust so it is important that the user verify that the data is being read in properly. This is especially important if versions of MD other than 6/93 are used.

3.3 CTA METHODOLOGY

The CTA program reads data from the MD output file (FOR006.DAT) and inserts it into two arrays. A one dimensional array, ALPHA, contains the angle of attack sweep data and the three dimensional BC array contains all other input data. The first and second dimension of BC (row and column) are listed in the table below. The first column contains general information about that dataset while the next 15 columns contain the coefficients over the AOA sweep contained in the ALPHA array. The third dimension of BC is the dataset number. Up to 20 angles of attack, 16 Mach numbers and 100 data sets can be handled with the current program dimensions.

Table 4 BC Array.

R\C	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16
		C_N	C_m	C_A	C_Y	C_n	C_l	$C_{N\alpha}$	$C_{m\alpha}$	$C_{Y\beta}$	$C_{n\beta}$	$C_{l\beta}$	Xcp	C_{Nq}	$C_{n\dot{\alpha}}$	$C_{mq}+C_{m\dot{\alpha}}$
1	NALP	@α_1	@α_1	...	...											
2	NMACH	@α_2	@α_2	...	...											
3	MACH	@α_3	@α_3	...	...											
4	ALT	@α_4	@α_4	...	...											
5	RE	...	...													
6	BETA	...	...													
7	PHI	...	...													
8	SREF															
9	LREF															
10	XCG															
11	YCG															
12	δ_1															
13	δ_2															
14	δ_3															
15	δ_4															
16																

MD calculates all the coefficients listed in Table 1 except C_{l_p}, so an alternate method of calculating this coefficient is needed. A method which predicts the roll damping derivatives of cruciform-tailed missiles was found in reference 6, using the following equation:

$$C_{lp} = -2.15(Y_{CENT} / l_{REF})C_{l_{\delta p}}$$

It is based on empirical correlation of experimental data for several cruciform-tailed missiles at Mach numbers from 0 to 4.0. Y_{cent}, the radial distance from body centerline to centroid of area of exposed tail fin, is input in the CTA.DAT file.

4. DATA COMPARISONS

This section will compare the CTA generated PA aerodynamic coefficients to those found in the Rockwell reports (Ref. 2 & Ref. 3). Rockwell used the ALSAC computer code and Rockwell International Missile Drag and Aerodynamic Manuals to generate their aerodynamic data. Wind tunnel data in both PA and CA format would be the best way to verify that the conversion process in the CTA code is working properly since differences can be attributed to the various methods used by Rockwell and by MD.

Figure 4 compares C_N' versus angle of attack at two Mach numbers. The CTA data has a smaller slope than the Rockwell data and shows little variation with Mach number at angles of attack less than 14°.

Figure 5 compares $\Delta C'_{N,\phi'}$ versus angle of attack at two Mach numbers. Rockwell predicts higher values over the angle of attack range at the lower Mach number but the data comes much closer to matching at the higher Mach number.

Figure 6 compares C_m' versus angle of attack at two Mach numbers. The Rockwell data shows very little variation with Mach number, the CTA data only slightly more. The slope trends are also different but the overall agreement is quite good.

Figure 7 shows $\Delta C'_{m,\phi'}$ versus angle of attack at two Mach numbers. The slope of the CTA data decreases at a much higher rate with increasing angle of attack than does the slope of the Rockwell data.

Figure 8 shows $\Delta C'_{Y,\phi'}$ versus angle of attack. Although the overall trends are similar, the Rockwell method predicts larger magnitude values at the low Mach number and smaller magnitude values at high Mach number than does the CTA method.

Figure 9 compares $\Delta C'_{n,\phi'}$ versus angle of attack at two Mach numbers. CTA predicts higher values as angle of attack increases and shows a larger variation with Mach number than the Rockwell data.

Figure 10 shows C_{A_o} versus Mach number. The data matches closely in the transonic region but diverges in the subsonic and supersonic region.

Figure 11 shows $C_{A_{\alpha'}}$ versus Mach number. Although the data fluctuates about the same region, the trends are similar only at the higher Mach numbers.

Figure 12 compares $C_{A_{\delta^2 eff}}$ versus Mach number. The Rockwell data shows no variation with Mach number while CTA predicts a curve similar in shape to the C_{A_o} versus Mach number curve found in Figure 10, with a transonic peak.

Figure 13 shows $C'_{N_{\delta q}}$ versus Mach number. The extremes of Mach number have similar values but the transonic region is predicted to be greater with the CTA method.

Figure 14 shows $C'_{m_{\delta q}}$ versus Mach number. The results are similar to those in Figure 12, with the CTA transonic peak of greater magnitude.

Figure 15 shows C'_{m_q} versus Mach number. The supersonic trends and magnitudes match very well but the transonic magnitudes differ by a substantial margin.

Figure 16 shows $C_{l,\phi'_{\alpha^2}}$ versus Mach number. The overall trends are similar but the magnitudes are not.

Figure 17 compares $C_{l_{\delta p}}$ versus Mach number. Again, the trends are similar but the magnitudes differ substantially.

Figure 18 shows C_{l_p} versus Mach number. The magnitudes differ substantially but are similar in that both changing little across the Mach number range.

5. CONCLUSIONS

The data comparison shows definite differences in results although general trends were often similar. These differences were expected given that both data are estimations utilizing different methods. For a true test of CTA, "verified" aerodynamic data in both CA and PA format is required since CTA only converts MD data to a different format.

The ability to quickly and accurately convert MD aerodynamic data to the PA format is important if PA equations are used in a flyout simulation. Although more testing is in order, the CTA code appears to accomplish the conversion process.

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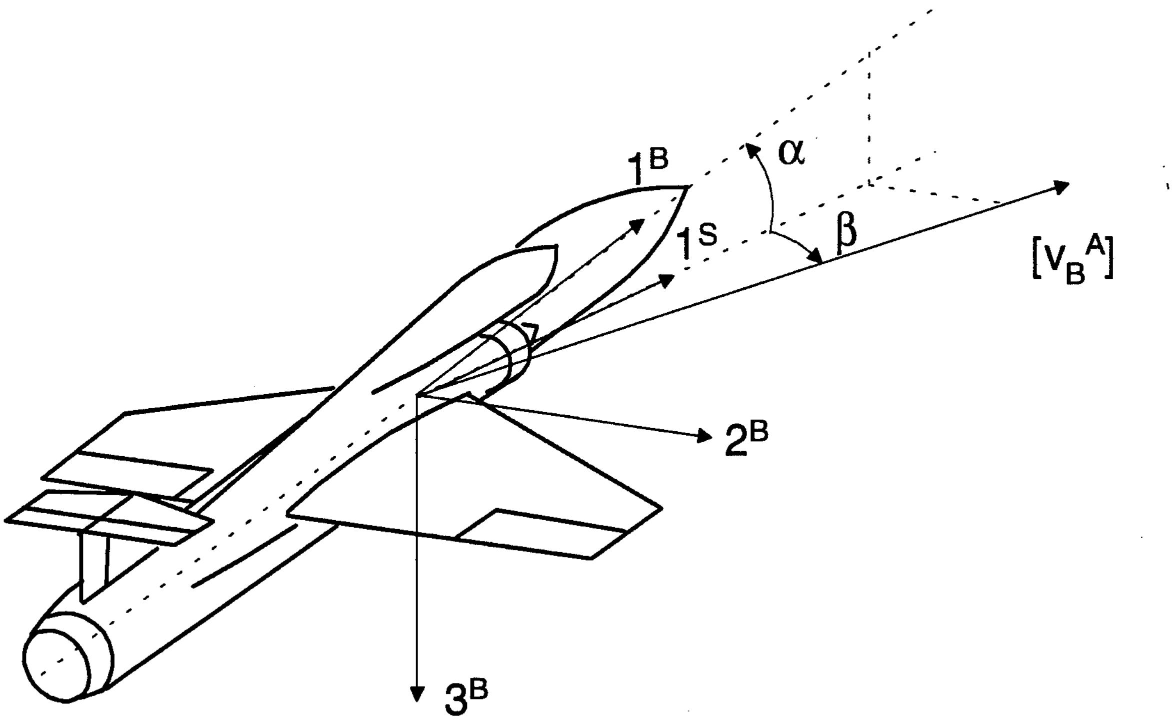

Figure 1. Aircraft Incidence Angles, α and β.

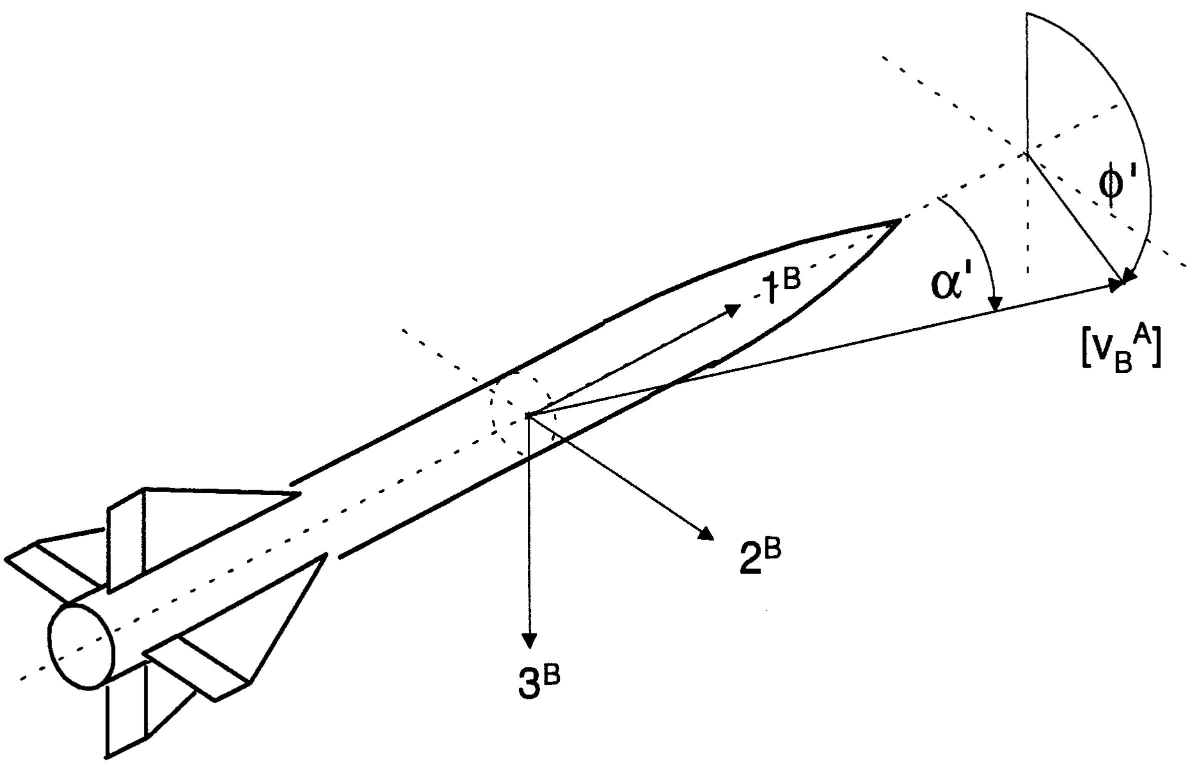

Figure 2. Missile Incidence Angles α' and φ'.

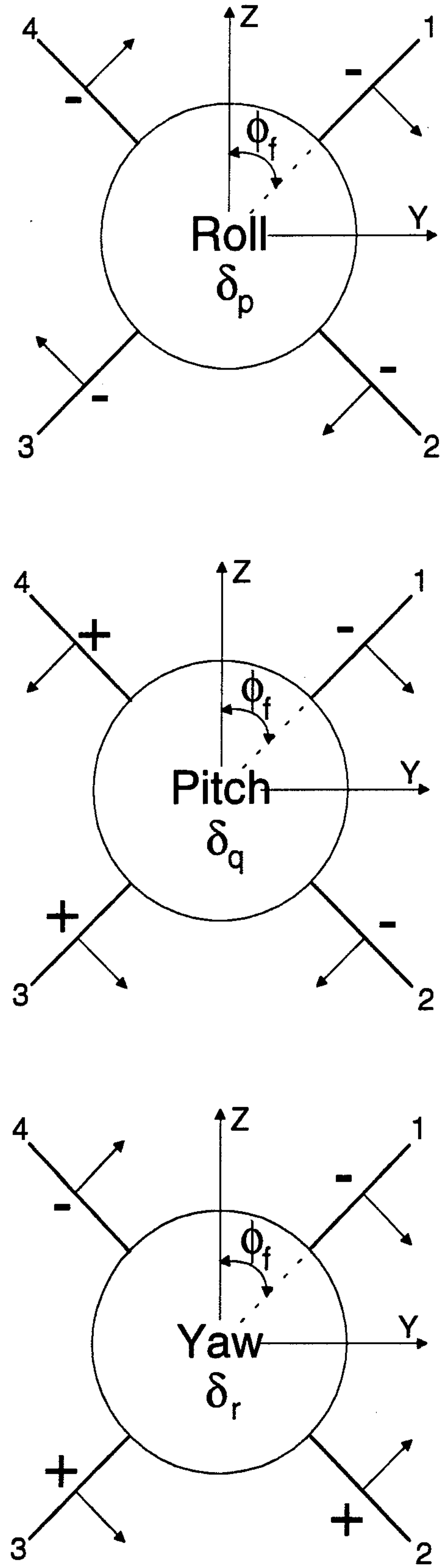

Figure 3. Definitions of Pitch, Yaw and Roll Control.

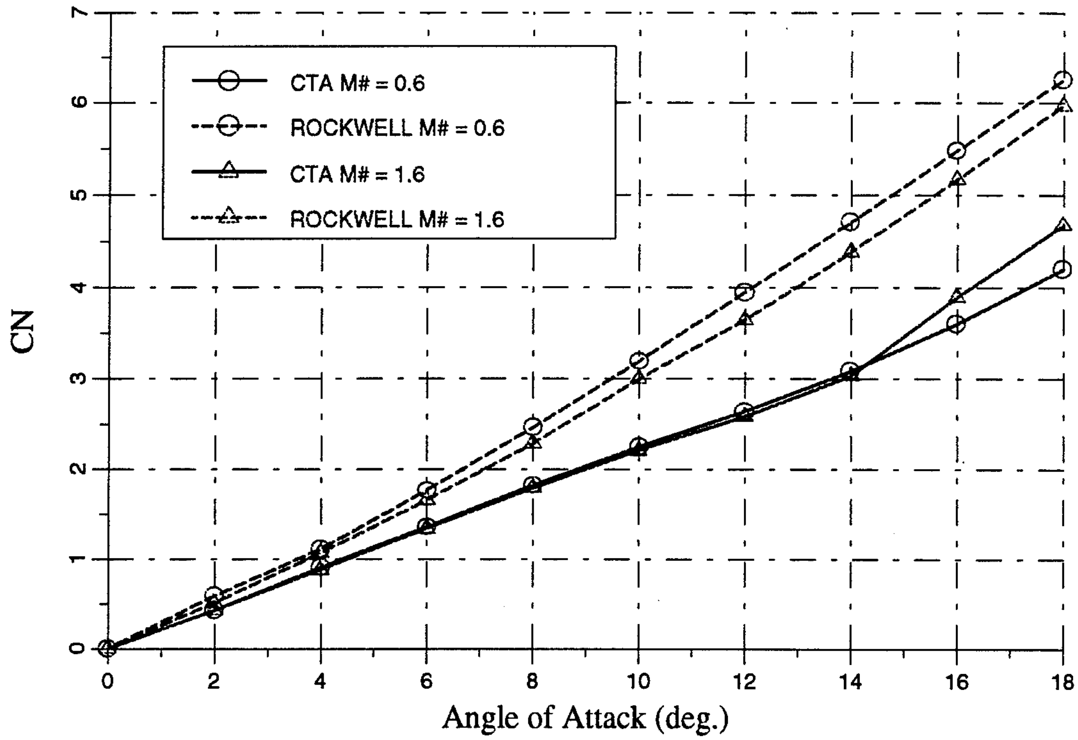

Figure 4. C_N' **versus Angle of Attack.**

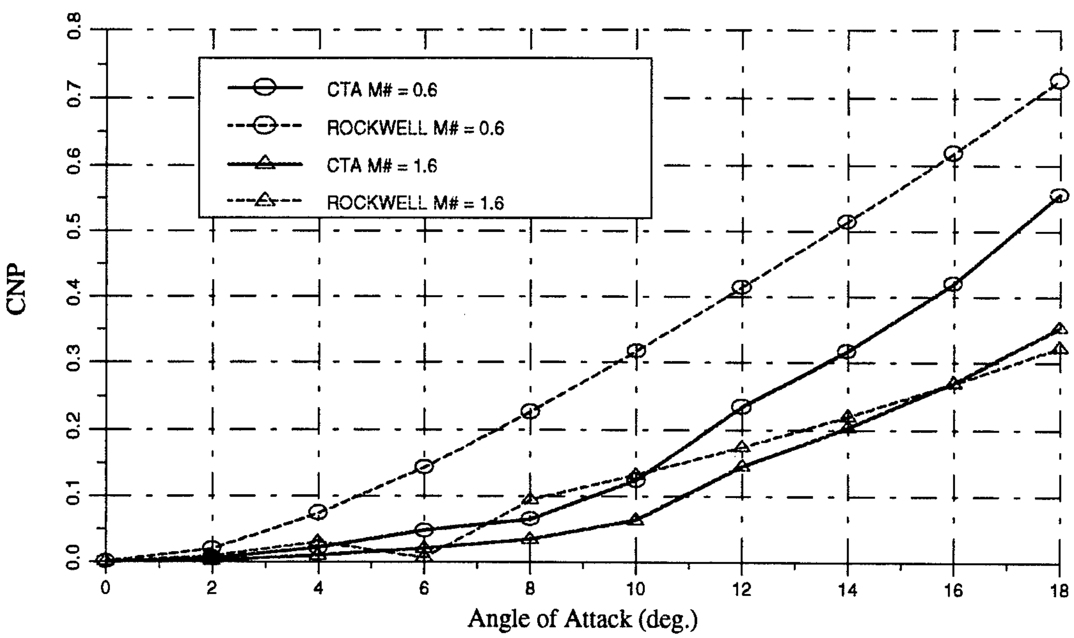

Figure 5. $\Delta C_{N,\phi'}'$ **versus Angle of Attack.**

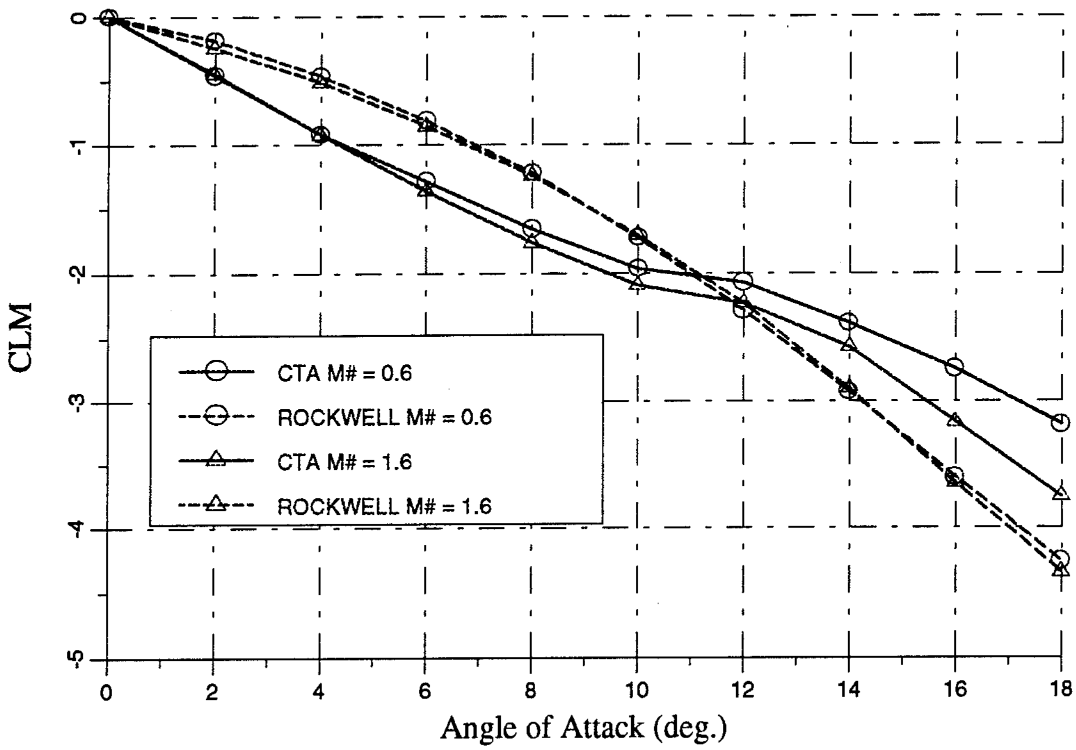

Figure 6. C'_m **versus Angle of Attack.**

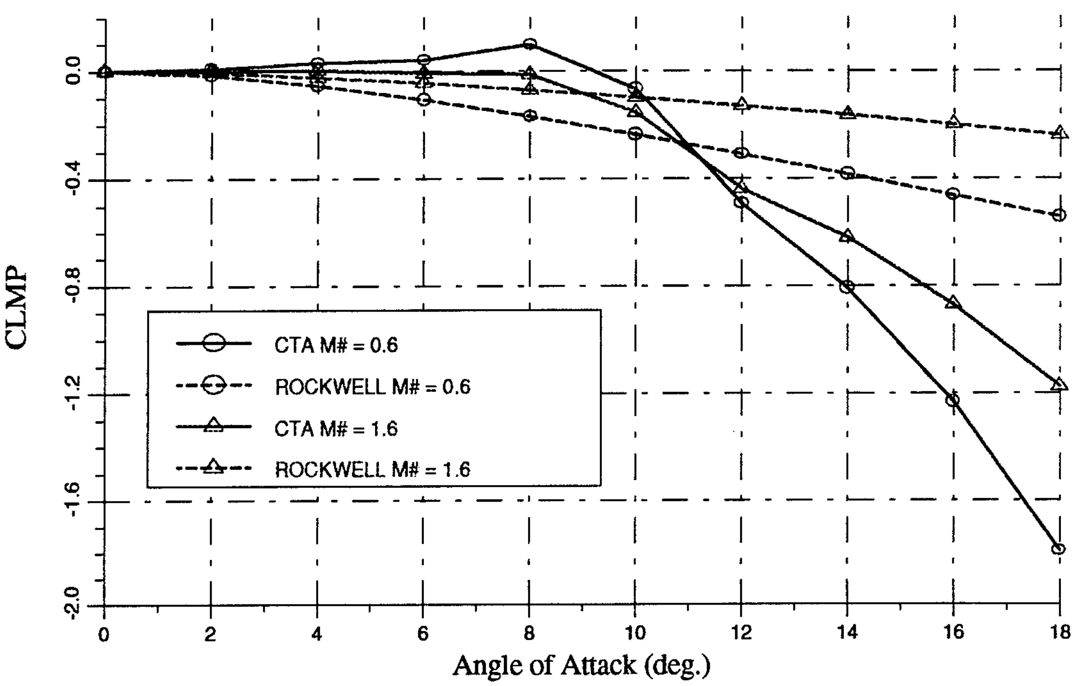

Figure 7. $\Delta C'_{m,\phi'}$ **versus Angle of Attack.**

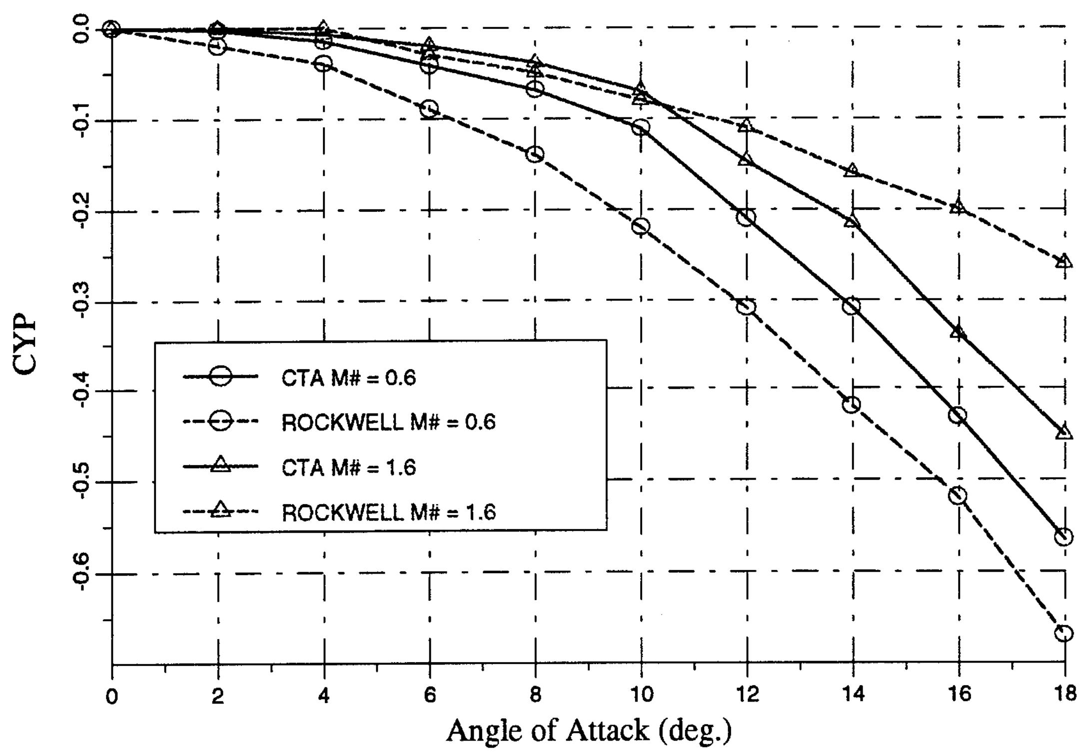

Figure 8. $\Delta C'_{Y,\phi'}$ **versus Angle of Attack.**

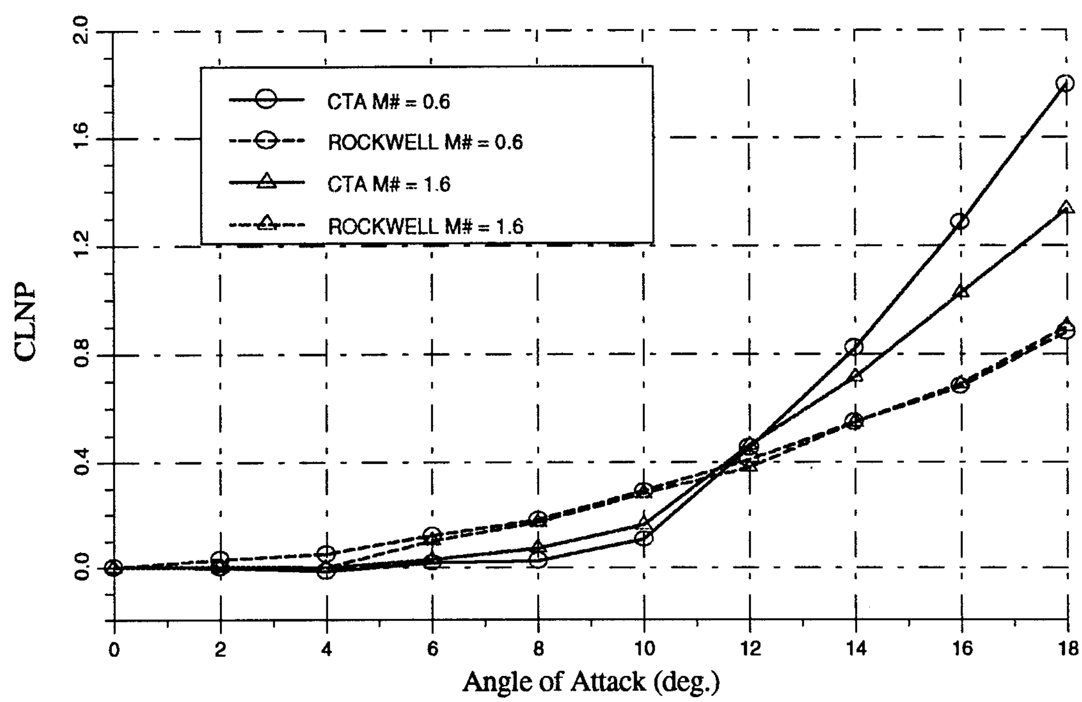

Figure 9. $\Delta C'_{n,\phi'}$ **versus Angle of Attack.**

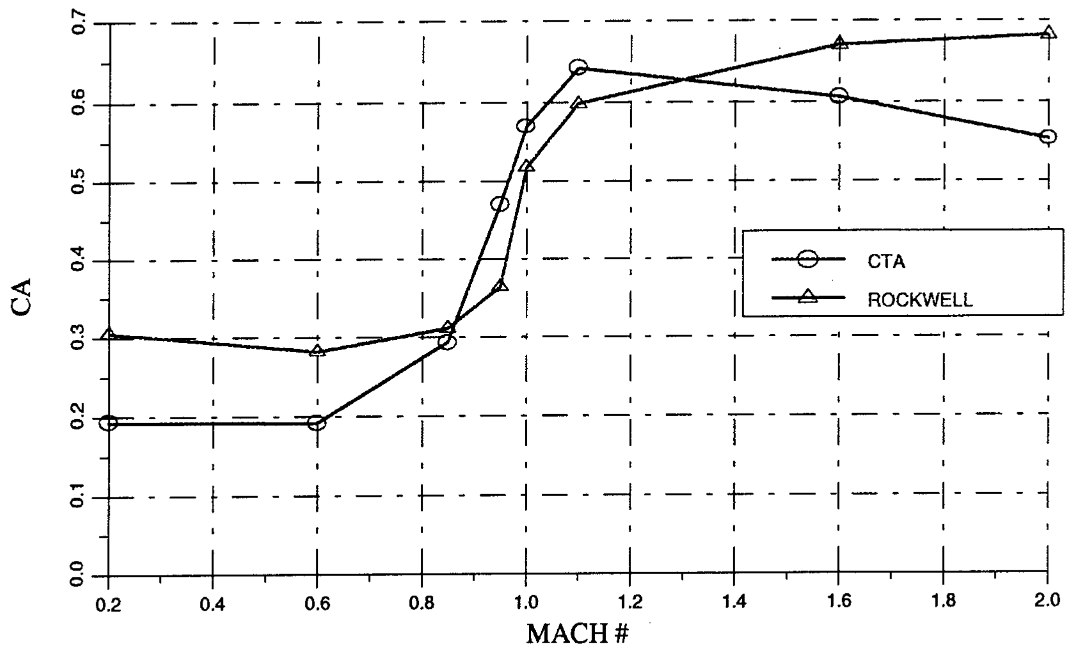

Figure 10. C_{A_o} **versus Mach Number.**

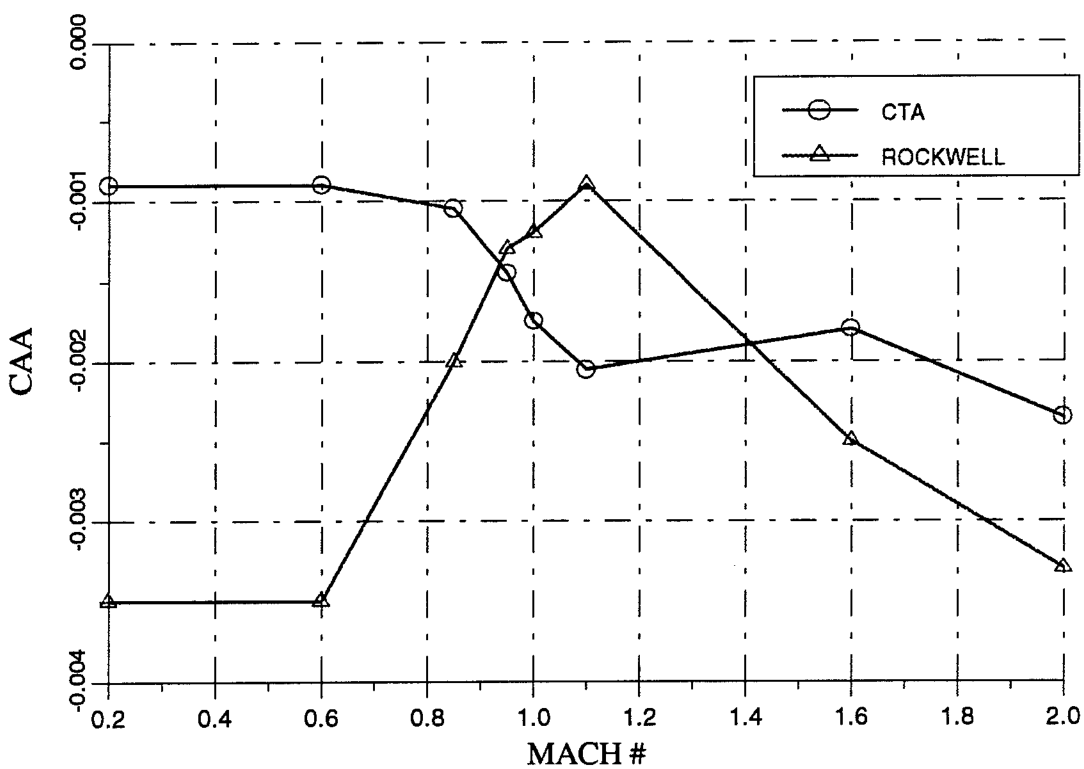

Figure 11. $C_{A_{\alpha'}}$ **versus Mach Number.**

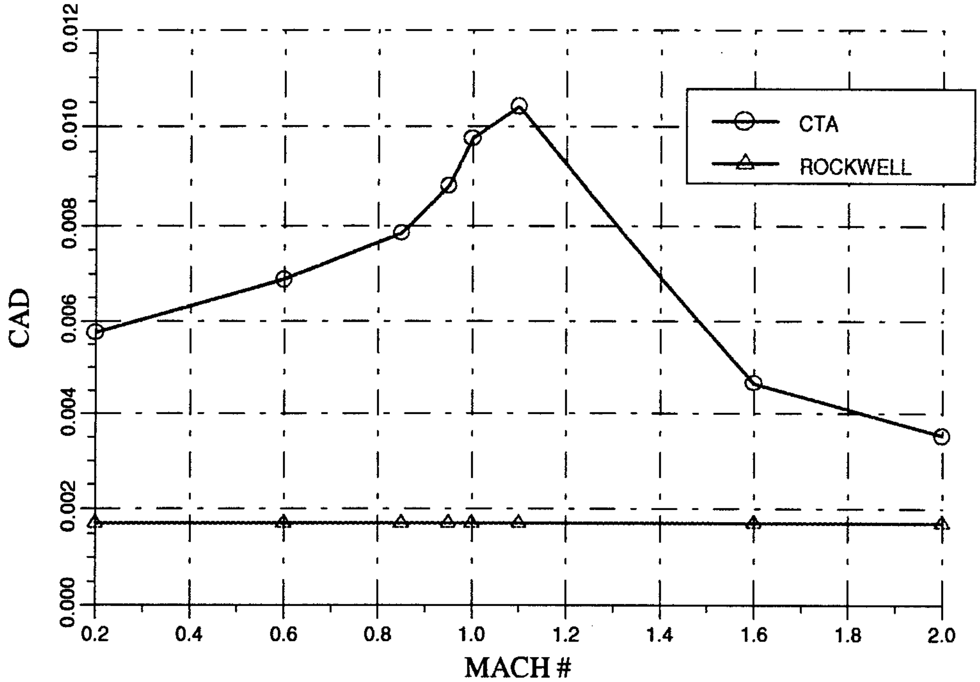

Figure 12. $C_{A_{\delta^2 eff}}$ **versus Mach Number.**

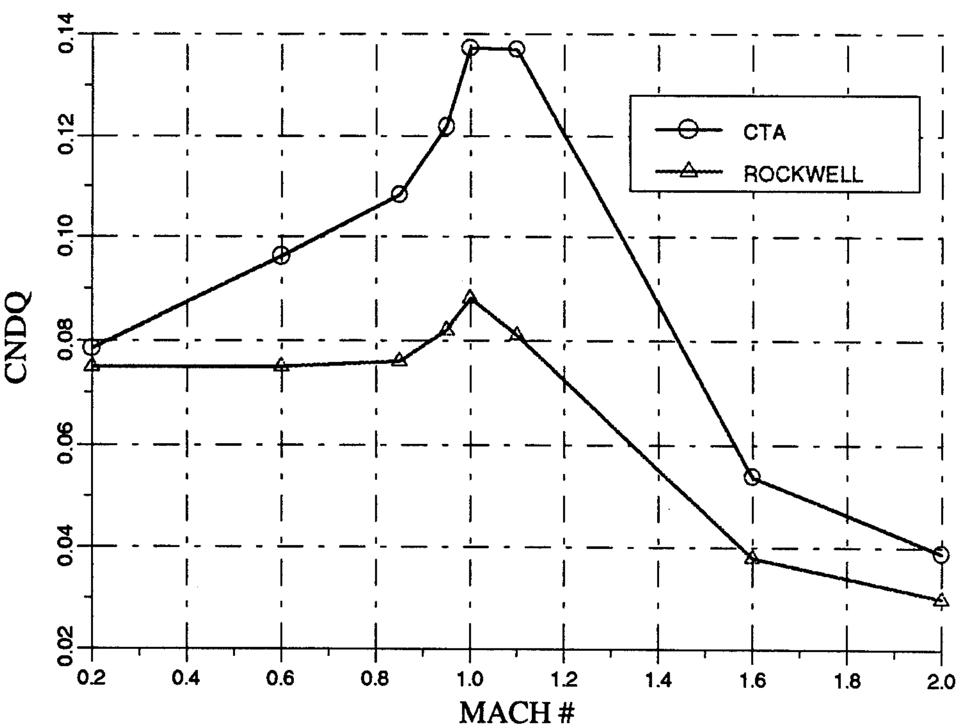

Figure 13. $C'_{N_{\delta q}}$ **versus Mach Number.**

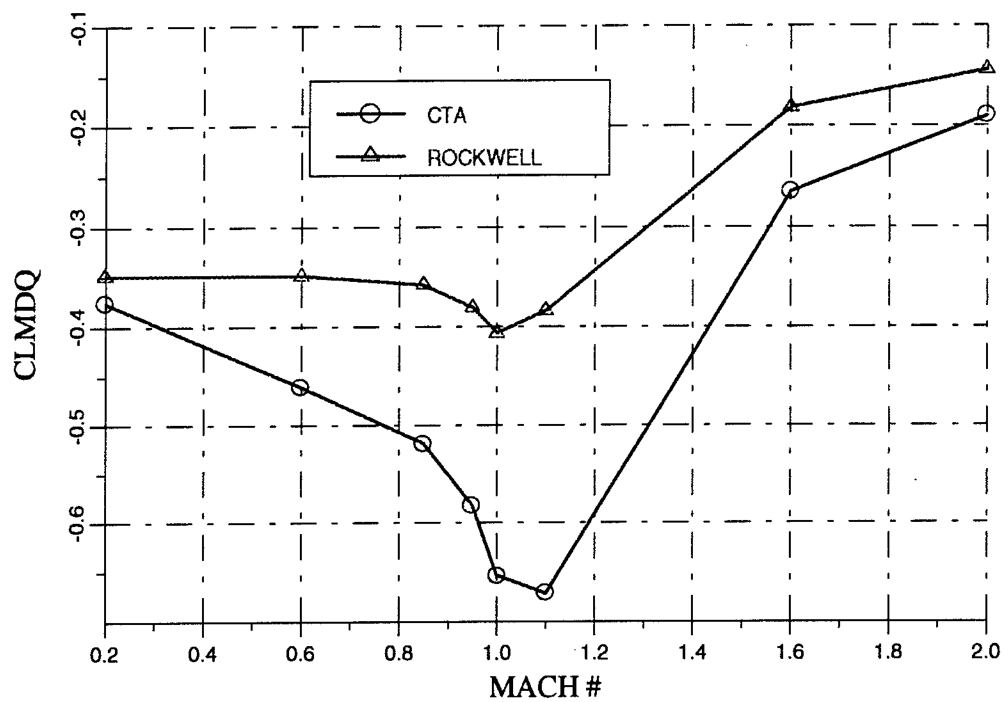

Figure 14. $C'_{m_{\delta q}}$ **versus Mach Number.**

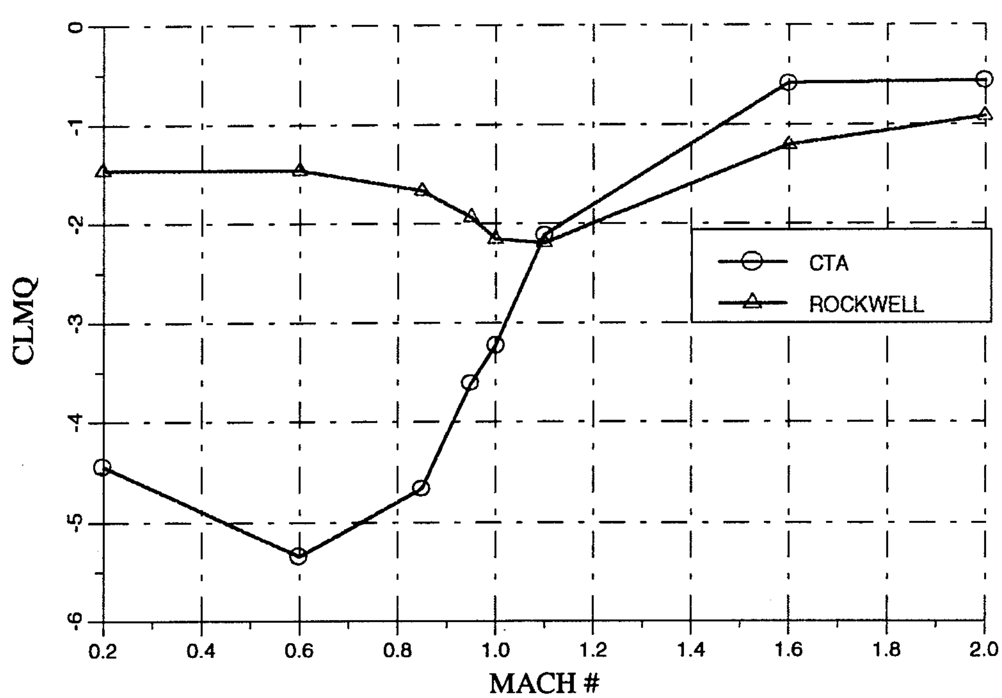

Figure 15. C'_{m_q} **versus Mach Number.**

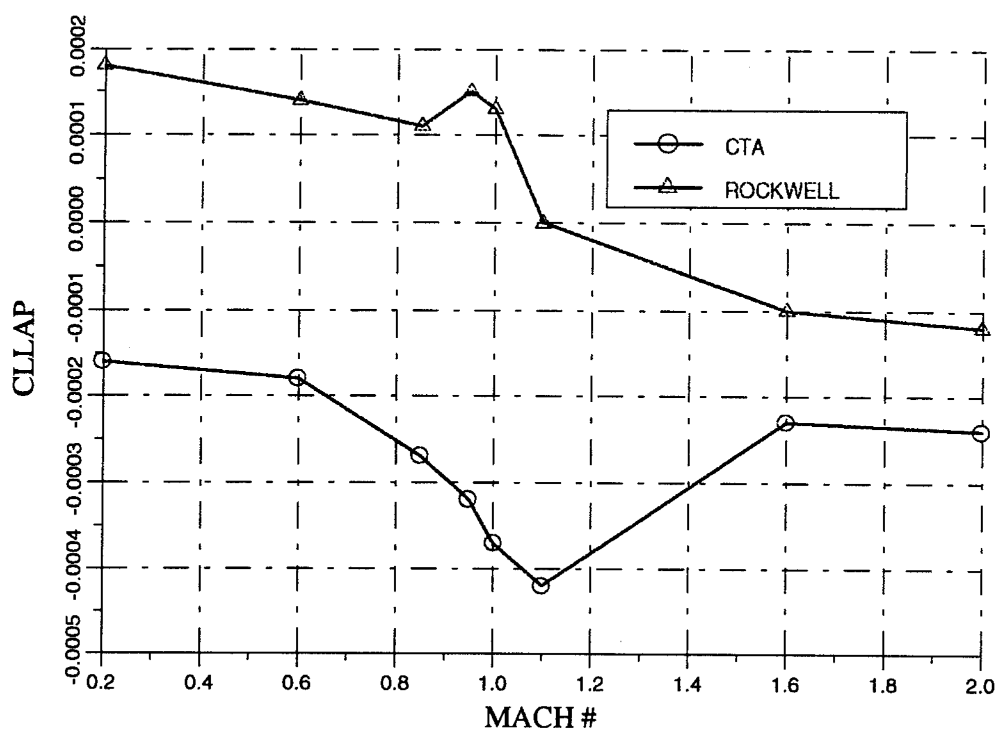

Figure 16. $C_{l,\phi'_{\alpha}2}$ **versus Mach Number.**

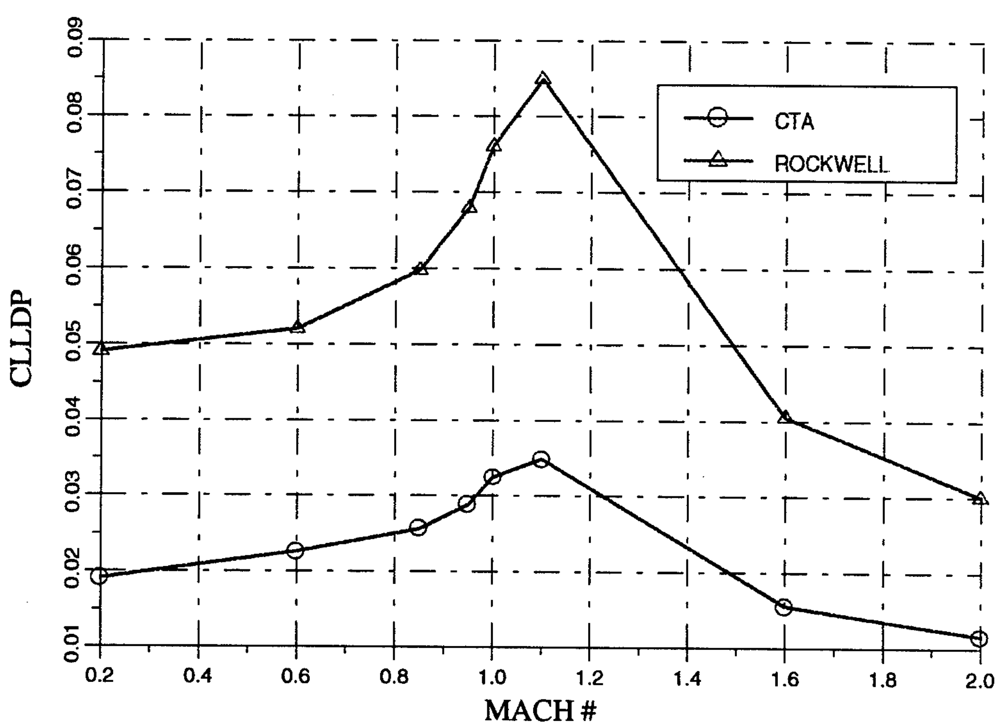

Figure 17. $C_{l_{\delta p}}$ **versus Mach Number.**

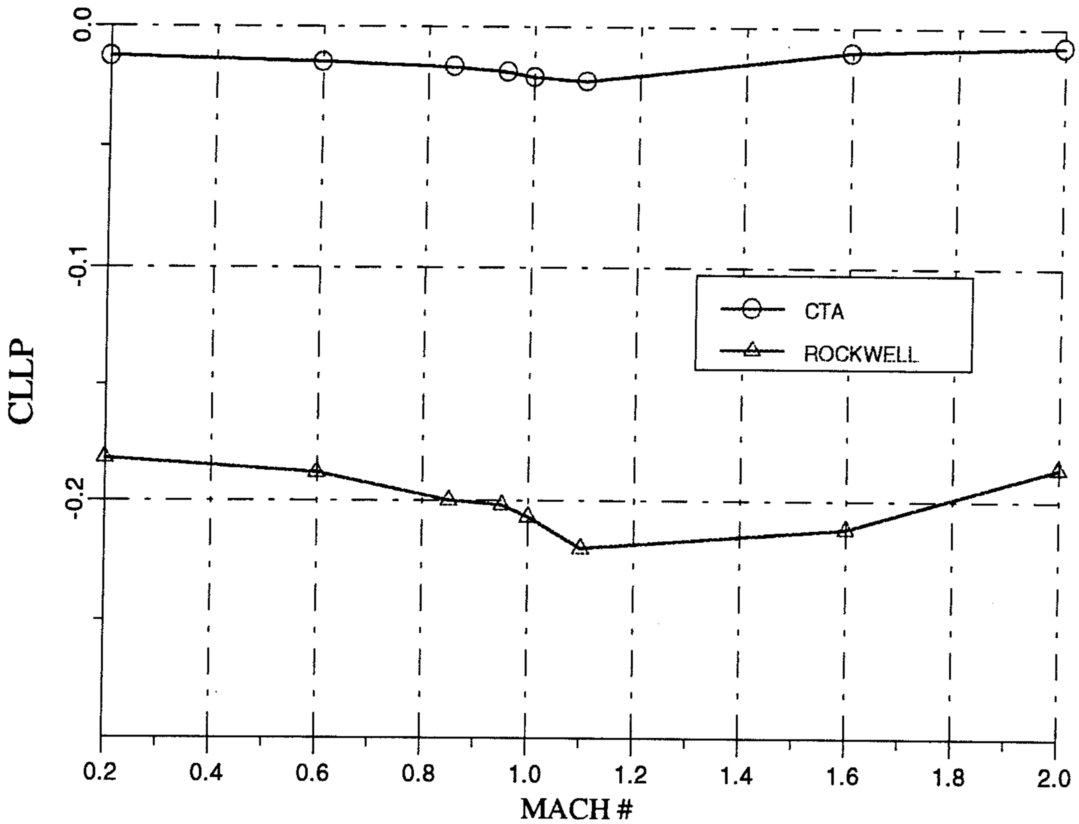

Figure 18. C_{l_p} **versus Mach Number.**

APPENDIX A - FOR005.DAT

```
 CASEID ROCKPRN
  $FLTCON  NMACH=08., MACH=0.2,0.6,0.85,0.95,1.0,1.1,1.6,2.0,
           NALPHA=11., ALPHA=0.,2.,4.,6.,8.,10.,12.,14.,16.,18.,20.,
           ALT=0000.0,PHI=0.0,$
  $REFQ    XCG=5.4583, ZCG=0., LREF=1.25, SREF=1.23,
           BLAYER=NATURAL, RHR=125., SCALE=1.0,$
  $AXIBOD
           TNOSE   =  CONICAL,
           LNOSE   =  2.0917,
           DNOSE   =  1.0833,
           BNOSE   =  0.375,
           XO      =  0.0,
           LCENTR  =  8.5,
           DCENTR  =  1.25,
           TAFT    =  CONICAL,
           LAFT    =  1.4483,
           DAFT    =  0.6021,
           DEXIT   =  0.0,$
  $FINSET1
           SECTYP  =  HEX,
           ZUPPER  =  0.008,0.0183,0.0432,
           LMAXU   =  0.7,0.3,0.25,
           LFLATU  =  0.0,0.0,0.0,
           SSPAN   =  0.0, 0.5833,1.1667,
           CHORD   =  3.9167, 1.2833,0.3667,
           XLE     =  4.0155,
           SWEEP   =  78.5, 57.4,
           STA     =  0., 0.,0.,
           NPANEL  =  4.,
           PHIF    =  45., 135., 225., 315.,$
  $FINSET2
           SECTYP  =  HEX,
           ZUPPER  =  0.0469,0.0156,
           LMAXU   =  0.33,0.25,
           LFLATU  =  0.0,0.0,
           SSPAN   =  0., 0.9167,
           CHORD   =  0.6667, 0.3333,
           XLE     =  11.1917,11.525,
*           SWEEP   =  20., 0.,
*           STA     =  0., 1.,
           NPANEL  =  4.,
           PHIF    =  45., 135., 225., 315.,$
  $DEFLCT
           DELTA2  =  0., 0., 0., 0.,
           XHINGE  =  6.7209,11.4584,$
 DIM FT
 DERIV DEG
 SOSE
 DAMP
 SAVE
 NEXT CASE
 CASEID ROCKII - PHI 22.5
  $FLTCON  PHI=22.5,$
 DERIV DEG
 DAMP
```

```
SAVE
NEXT CASE
CASEID ROCKII - PHI 45
 $FLTCON  PHI=45.0,$
DERIV DEG
DAMP
SAVE
NEXT CASE
CASEID ROCKII - dp = 5 deg
 $FLTCON  PHI=0.0,$
 $DEFLCT        DELTA2  =  -5., -5., -5., -5.,$
DERIV DEG
DAMP
SAVE
NEXT CASE
CASEID ROCKII - dq = 5 deg
 $DEFLCT        DELTA2  = -5., -5., 5., 5.,$
DERIV DEG
DAMP
SAVE
NEXT CASE
CASEID ROCKII - dr = 5 deg
 $DEFLCT        DELTA2  =  -5., 5., 5., -5.,$
DERIV DEG
DAMP
SAVE
NEXT CASE
CASEID ROCKII - deff = 5 deg
 $DEFLCT        DELTA2  = -5., -5., 5., 5.,$
DERIV DEG
DAMP
SAVE
NEXT CASE
CASEID ROCKII - dp = 5 deg, phi = 22.5 deg
 $FLTCON  PHI=22.5,$
 $DEFLCT        DELTA2  =  -5., -5., -5., -5.,$
DERIV DEG
DAMP
NEXT CASE
```

APPENDIX B - FOR006.DAT

```
                            THE USAF AUTOMATED MISSILE DATCOM * REV  6/93 *
                            AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
                                 CONERR - INPUT ERROR CHECKING

ERROR CODES - N* DENOTES THE NUMBER OF OCCURENCES OF EACH ERROR

A - UNKNOWN VARIABLE NAME

B - MISSING EQUAL SIGN FOLLOWING VARIABLE NAME

C - NON-ARRAY VARIABLE HAS AN ARRAY ELEMENT DESIGNATION - (N)

D - NON-ARRAY VARIABLE HAS MULTIPLE VALUES ASSIGNED

E - ASSIGNED VALUES EXCEED ARRAY DIMENSION

F - SYNTAX ERROR

  ******************************  INPUT DATA CARDS  ******************************

 1 CASEID ROCKPRN
 2  $FLTCON  NMACH=08., MACH=0.2,0.6,0.85,0.95,1.0,1.1,1.6,2.0,
 3           NALPHA=11., ALPHA=0.,2.,4.,6.,8.,10.,12.,14.,16.,18.,20.,
 4           ALT=0000.0,PHI=0.0,$
 5  $REFQ    XCG=5.4583, ZCG=0., LREF=1.25, SREF=1.23,
 6           BLAYER=NATURAL, RHR=125., SCALE=1.0,$          ** SUBSTITUTING NUMERIC FOR NAME NATURAL
 7  $AXIBOD
 8           TNOSE   =  CONICAL,                            ** SUBSTITUTING NUMERIC FOR NAME CONICAL
 9           LNOSE   =  2.0917,
10           DNOSE   =  1.0833,
11           BNOSE   =  0.375,
12           XO      =  0.0,
13           LCENTR  =  8.5,
14           DCENTR  =  1.25,
15           TAFT    =  CONICAL,                            ** SUBSTITUTING NUMERIC FOR NAME CONICAL
16           LAFT    =  1.4483,
17           DAFT    =  0.6021,
18           DEXIT   =  0.0,$
19  $FINSET1
20           SECTYP  =  HEX,                                ** SUBSTITUTING NUMERIC FOR NAME HEX
21           ZUPPER  =  0.008,0.0183,0.0432,
22           LMAXU   =  0.7,0.3,0.25,
23           LFLATU  =  0.0,0.0,0.0,
24           SSPAN   =  0.0, 0.5833,1.1667,
25           CHORD   =  3.9167, 1.2833,0.3667,
26           XLE     =  4.0155,
27           SWEEP   =  78.5, 57.4,
28           STA     =  0., 0.,0.,
29           NPANEL  =  4.,
30           PHIF    =  45., 135., 225., 315.,$
31  $FINSET2
32           SECTYP  =  HEX,                                ** SUBSTITUTING NUMERIC FOR NAME HEX
33           ZUPPER  =  0.0469,0.0156,
34           LMAXU   =  0.33,0.25,
35           LFLATU  =  0.0,0.0,
36           SSPAN   =  0., 0.9167,
37           CHORD   =  0.6667, 0.3333,
38           XLE     =  11.1917,11.525,
39 *          SWEEP   =  20., 0.,
```

```
*              STA      =  0., 1.,
              NPANEL   =  4.,
              PHIF     =  45., 135., 225., 315.,$
 $DEFLCT
              DELTA2   =  0., 0., 0., 0.,
              XHINGE   =  6.7209,11.4584,$
DIM FT
DERIV DEG
SOSE
DAMP
SAVE
NEXT CASE
CASEID ROCKII - PHI 22.5
 $FLTCON  PHI=22.5,$
DERIV DEG
DAMP
SAVE
NEXT CASE
CASEID ROCKII - PHI 45
 $FLTCON  PHI=45.0,$
DERIV DEG
DAMP
SAVE
NEXT CASE
CASEID ROCKII - dp = 5 deg
 $FLTCON  PHI=0.0,$
 $DEFLCT        DELTA2  =  -5., -5., -5., -5.,$
DERIV DEG
DAMP
SAVE
NEXT CASE
CASEID ROCKII - dq = 5 deg
 $DEFLCT        DELTA2  = -5., -5., 5., 5.,$
DERIV DEG
DAMP
SAVE
NEXT CASE
CASEID ROCKII - dr = 5 deg
 $DEFLCT        DELTA2  =  -5., 5., 5., -5.,$
DERIV DEG
DAMP
SAVE
NEXT CASE
CASEID ROCKII - deff = 5 deg
 $DEFLCT        DELTA2  = -5., -5., 5., 5.,$
DERIV DEG
DAMP
SAVE
NEXT CASE
CASEID ROCKII - dp = 5 deg, phi = 22.5 deg
 $FLTCON  PHI=22.5,$
 $DEFLCT        DELTA2  =  -5., -5., -5., -5.,$
DERIV DEG
DAMP
NEXT CASE
```

```
FOLLOWING ARE THE CARDS INPUT FOR THIS CASE

CASEID ROCKPRN
 $FLTCON  NMACH=08., MACH=0.2,0.6,0.85,0.95,1.0,1.1,1.6,2.0,
           NALPHA=11., ALPHA=0.,2.,4.,6.,8.,10.,12.,14.,16.,18.,20.,
           ALT=0000.0,PHI=0.0,$
 $REFQ    XCG=5.4583, ZCG=0., LREF=1.25, SREF=1.23,
           BLAYER=1., RHR=125., SCALE=1.0,$
 $AXIBOD
           TNOSE    =  0.,
           LNOSE    =  2.0917,
           DNOSE    =  1.0833,
           BNOSE    =  0.375,
           XO       =  0.0,
           LCENTR   =  8.5,
           DCENTR   =  1.25,
           TAFT     =  0.,
           LAFT     =  1.4483,
           DAFT     =  0.6021,
           DEXIT    =  0.0,$
 $FINSET1
           SECTYP   =  0.,
           ZUPPER   =  0.008,0.0183,0.0432,
           LMAXU    =  0.7,0.3,0.25,
           LFLATU   =  0.0,0.0,0.0,
           SSPAN    =  0.0, 0.5833,1.1667,
           CHORD    =  3.9167, 1.2833,0.3667,
           XLE      =  4.0155,
           SWEEP    =  78.5, 57.4,
           STA      =  0., 0.,0.,
           NPANEL   =  4.,
           PHIF     =  45., 135., 225., 315.,$
 $FINSET2
           SECTYP   =  0.,
           ZUPPER   =  0.0469,0.0156,
           LMAXU    =  0.33,0.25,
           LFLATU   =  0.0,0.0,
           SSPAN    =  0., 0.9167,
           CHORD    =  0.6667, 0.3333,
           XLE      =  11.1917,11.525,
*           SWEEP    =  20., 0.,
*           STA      =  0., 1.,
           NPANEL   =  4.,
           PHIF     =  45., 135., 225., 315.,$
 $DEFLCT
           DELTA2   =  0., 0., 0., 0.,
           XHINGE   =  6.7209,11.4584,$
DIM FT
DERIV DEG
SOSE
DAMP
SAVE
NEXT CASE

THE BOUNDARY LAYER IS ASSUMED TO DEVELOP NATURALLY OVER ALL COMPONENTS OF THE CONFIGURATION
```

THE INPUT UNITS ARE IN FEET, THE SCALE FACTOR IS 1.0000

THE USAF AUTOMATED MISSILE DATCOM * REV 6/93 *
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
ROCKPRN
STATIC AERODYNAMICS FOR BODY-FIN SET 1 AND 2

CASE 1
PAGE 2

FLIGHT CONDITIONS								REFERENCE DIMENSIONS				
MACH NUMBER	ALTITUDE	VELOCITY	PRESSURE	TEMPERATURE	REYNOLDS NUMBER	SIDESLIP ANGLE	ROLL ANGLE	REF. AREA	REF. LENGTH LONG.	REF. LENGTH LAT.	MOMENT REF. CENTER LONG.	MOMENT REF. CENTER VERTICAL
	FT	FT/SEC	LB/FT**2	DEG R	1/FT	DEG	DEG	FT**2	FT	FT	FT	FT
0.20	0.00	223.25	2.116E+03	518.67	1.414E+06	0.00	0.00	1.230	1.250	1.250	5.458	0.000

	LONGITUDINAL			LATERAL DIRECTIONAL			DERIVATIVES (PER DEGREE) LONGITUDINAL		DERIVATIVES (PER DEGREE) LATERAL DIRECTIONAL		
ALPHA	CN	CM	CA	CY	CLN	CLL	CNA	CMA	CYB	CLNB	CLLB
0.00	0.000	0.000	0.193	0.000	0.000	0.000	1.978E-01	-2.198E-01	-2.062E-01	2.211E-01	-6.985E-10
2.00	0.423	-0.445	0.193	0.000	0.000	0.000	2.248E-01	-2.251E-01	-2.178E-01	2.186E-01	1.470E-04
4.00	0.899	-0.900	0.193	0.000	0.000	0.000	2.373E-01	-1.748E-01	-2.388E-01	2.138E-01	6.995E-04
6.00	1.372	-1.146	0.192	0.000	0.000	0.000	2.348E-01	-1.388E-01	-2.612E-01	2.437E-01	-2.160E-03
8.00	1.838	-1.456	0.191	0.000	0.000	0.000	2.231E-01	-1.438E-01	-2.697E-01	2.432E-01	-2.695E-03
10.00	2.264	-1.721	0.190	0.000	0.000	0.000	2.239E-01	-1.781E-01	-2.715E-01	2.436E-01	-3.642E-03
12.00	2.734	-2.169	0.188	0.000	0.000	0.000	2.323E-01	-1.924E-01	-2.762E-01	2.620E-01	-4.077E-03
14.00	3.193	-2.491	0.185	0.000	0.000	0.000	2.300E-01	-1.547E-01	-2.874E-01	3.134E-01	-8.020E-03
16.00	3.654	-2.788	0.182	0.000	0.000	0.000	2.359E-01	-1.557E-01	-2.929E-01	3.484E-01	-1.223E-02
18.00	4.137	-3.114	0.179	0.000	0.000	0.000	2.441E-01	-1.400E-01	-2.928E-01	3.803E-01	-1.809E-02
20.00	4.631	-3.348	0.175	0.000	0.000	0.000	2.495E-01	-9.402E-02	-2.969E-01	4.172E-01	-2.321E-02

ALPHA	CL	CD	CL/CD	X-C.P.
0.00	0.000	0.193	0.000	-1.111
2.00	0.416	0.208	2.001	-1.053
4.00	0.884	0.255	3.466	-1.001
6.00	1.344	0.334	4.019	-0.835
8.00	1.794	0.445	4.030	-0.792
10.00	2.197	0.580	3.787	-0.760
12.00	2.635	0.752	3.503	-0.793
14.00	3.054	0.952	3.207	-0.780
16.00	3.462	1.182	2.928	-0.763
18.00	3.879	1.449	2.677	-0.753
20.00	4.291	1.749	2.454	-0.723

PANEL DEFLECTION ANGLES (DEGREES)

FIN SET	FIN 1	FIN 2	FIN 3	FIN 4
1	0.00	0.00	0.00	0.00
2	0.00	0.00	0.00	0.00

THE USAF AUTOMATED MISSILE DATCOM * REV 6/93 *
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
ROCKPRN
BODY + 2 FIN SETS DYNAMIC DERIVATIVES

FLIGHT CONDITIONS								REFERENCE DIMENSIONS				
MACH NUMBER	ALTITUDE	VELOCITY	PRESSURE	TEMPERATURE	REYNOLDS NUMBER	SIDESLIP ANGLE	ROLL ANGLE	REF. AREA	REF. LENGTH LONG.	REF. LENGTH LAT.	MOMENT REF. CENTER LONG.	MOMENT REF. CENTER VERTICAL
	FT	FT/SEC	LB/FT**2	DEG R	1/FT	DEG	DEG	FT**2	FT	FT	FT	FT
0.20	0.00	223.25	2.116E+03	518.67	1.414E+06	0.00	0.00	1.230	1.250	1.250	5.458	0.000

DYNAMIC DERIVATIVES (PER DEGREE)

ALPHA	CNQ	CNAD	CMQ+CMAD
0.0	1.595E+00	6.042E-01	-4.44430E+00
2.0	1.756E+00	9.870E-01	-4.74070E+00
4.0	1.912E+00	1.341E+00	-5.02652E+00
6.0	2.034E+00	1.507E+00	-5.25561E+00
8.0	2.132E+00	1.543E+00	-5.44336E+00
10.0	2.230E+00	1.589E+00	-5.62679E+00
12.0	2.175E+00	1.571E+00	-5.30506E+00
14.0	1.937E+00	1.184E+00	-4.46262E+00
16.0	1.639E+00	3.709E-01	-3.54468E+00
18.0	1.416E+00	1.710E-01	-2.70570E+00
20.0	1.292E+00	6.847E-01	-1.99654E+00

THE USAF AUTOMATED MISSILE DATCOM * REV 6/93 *
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
ROCKPRN
STATIC AERODYNAMICS FOR BODY-FIN SET 1 AND 2

FLIGHT CONDITIONS								REFERENCE DIMENSIONS				
MACH NUMBER	ALTITUDE	VELOCITY	PRESSURE	TEMPERATURE	REYNOLDS NUMBER	SIDESLIP ANGLE	ROLL ANGLE	REF. AREA	REF. LENGTH LONG.	REF. LENGTH LAT.	MOMENT REF. CENTER LONG.	MOMENT REF. CENTER VERTICAL
	FT	FT/SEC	LB/FT**2	DEG R	1/FT	DEG	DEG	FT**2	FT	FT	FT	FT
0.60	0.00	669.76	2.116E+03	518.67	4.241E+06	0.00	0.00	1.230	1.250	1.250	5.458	0.000

	LONGITUDINAL			LATERAL DIRECTIONAL			DERIVATIVES (PER DEGREE) LONGITUDINAL		DERIVATIVES (PER DEGREE) LATERAL DIRECTIONAL		
ALPHA	CN	CM	CA	CY	CLN	CLL	CNA	CMA	CYB	CLNB	CLLB
0.00	0.000	0.000	0.192	0.000	0.000	0.000	2.055E-01	-2.303E-01	-2.121E-01	2.299E-01	-3.027E-09
2.00	0.431	-0.460	0.192	0.000	0.000	0.000	2.252E-01	-2.295E-01	-2.215E-01	2.245E-01	1.237E-04
4.00	0.901	-0.918	0.191	0.000	0.000	0.000	2.322E-01	-2.061E-01	-2.387E-01	2.147E-01	5.700E-04
6.00	1.359	-1.285	0.191	0.000	0.000	0.000	2.294E-01	-1.856E-01	-2.492E-01	2.157E-01	-1.972E-04
8.00	1.819	-1.661	0.190	0.000	0.000	0.000	2.208E-01	-1.697E-01	-2.540E-01	2.096E-01	-4.291E-04
10.00	2.243	-1.964	0.189	0.000	0.000	0.000	2.048E-01	-1.031E-01	-2.552E-01	2.120E-01	-1.814E-03
12.00	2.638	-2.074	0.188	0.000	0.000	0.000	2.099E-01	-1.061E-01	-2.683E-01	2.708E-01	-5.930E-03
14.00	3.082	-2.389	0.186	0.000	0.000	0.000	2.410E-01	-1.673E-01	-2.807E-01	3.317E-01	-1.098E-02
16.00	3.602	-2.743	0.183	0.000	0.000	0.000	2.800E-01	-1.990E-01	-2.910E-01	3.666E-01	-1.661E-02
18.00	4.203	-3.185	0.179	0.000	0.000	0.000	2.985E-01	-1.935E-01	-3.000E-01	3.994E-01	-2.354E-02
20.00	4.796	-3.518	0.174	0.000	0.000	0.000	2.950E-01	-1.388E-01	-3.117E-01	4.455E-01	-3.098E-02

ALPHA	CL	CD	CL/CD	X-C.P.
0.00	0.000	0.192	0.000	-1.121
2.00	0.424	0.207	2.051	-1.068
4.00	0.885	0.254	3.489	-1.019
6.00	1.332	0.332	4.013	-0.945
8.00	1.775	0.441	4.021	-0.913
10.00	2.176	0.576	3.780	-0.876
12.00	2.541	0.732	3.472	-0.786
14.00	2.946	0.926	3.181	-0.775
16.00	3.412	1.169	2.919	-0.762
18.00	3.942	1.469	2.683	-0.758
20.00	4.447	1.804	2.465	-0.733

PANEL DEFLECTION ANGLES (DEGREES)

FIN SET	FIN 1	FIN 2	FIN 3	FIN 4
1	0.00	0.00	0.00	0.00
2	0.00	0.00	0.00	0.00

THE USAF AUTOMATED MISSILE DATCOM * REV 6/93 *
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
ROCKPRN
BODY + 2 FIN SETS DYNAMIC DERIVATIVES

CASE 1
PAGE 5

FLIGHT CONDITIONS								REFERENCE DIMENSIONS				
MACH NUMBER	ALTITUDE FT	VELOCITY FT/SEC	PRESSURE LB/FT**2	TEMPERATURE DEG R	REYNOLDS NUMBER 1/FT	SIDESLIP ANGLE DEG	ROLL ANGLE DEG	REF. AREA FT**2	REF. LENGTH LONG. FT	REF. LENGTH LAT. FT	MOMENT REF. CENTER LONG. FT	MOMENT REF. CENTER VERTICAL FT
0.60	0.00	669.76	2.116E+03	518.67	4.241E+06	0.00	0.00	1.230	1.250	1.250	5.458	0.000

DYNAMIC DERIVATIVES (PER DEGREE)

ALPHA	CNQ	CNAD	CMQ+CMAD
0.0	1.737E+00	5.844E-01	-5.34555E+00
2.0	1.947E+00	7.277E-01	-5.92452E+00
4.0	2.122E+00	6.588E-01	-6.42175E+00
6.0	2.301E+00	6.285E-01	-6.92540E+00
8.0	2.520E+00	8.615E-01	-7.51738E+00
10.0	2.728E+00	1.053E+00	-8.08359E+00
12.0	2.889E+00	9.806E-01	-8.53951E+00
14.0	3.094E+00	1.209E+00	-9.09316E+00
16.0	3.337E+00	1.693E+00	-9.72667E+00
18.0	3.532E+00	1.919E+00	-1.02490E+01
20.0	3.724E+00	2.146E+00	-1.07628E+01

THE USAF AUTOMATED MISSILE DATCOM * REV 6/93 *
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
ROCKPRN
STATIC AERODYNAMICS FOR BODY-FIN SET 1 AND 2

FLIGHT CONDITIONS								REFERENCE DIMENSIONS				
MACH NUMBER	ALTITUDE	VELOCITY	PRESSURE	TEMPERATURE	REYNOLDS NUMBER	SIDESLIP ANGLE	ROLL ANGLE	REF. AREA	REF. LENGTH LONG.	REF. LENGTH LAT.	MOMENT REF. CENTER LONG.	MOMENT REF. CENTER VERTICAL
	FT	FT/SEC	LB/FT**2	DEG R	1/FT	DEG	DEG	FT**2	FT	FT	FT	FT
0.85	0.00	948.83	2.116E+03	518.67	6.008E+06	0.00	0.00	1.230	1.250	1.250	5.458	0.000

	LONGITUDINAL			LATERAL DIRECTIONAL			DERIVATIVES (PER DEGREE) LONGITUDINAL		DERIVATIVES (PER DEGREE) LATERAL DIRECTIONAL		
ALPHA	CN	CM	CA	CY	CLN	CLL	CNA	CMA	CYB	CLNB	CLLB
0.00	0.000	0.000	0.293	0.000	0.000	0.000	2.149E-01	-1.711E-01	-2.192E-01	1.705E-01	4.424E-09
2.00	0.441	-0.340	0.293	0.000	0.000	0.000	2.264E-01	-1.689E-01	-2.247E-01	1.640E-01	7.288E-05
4.00	0.906	-0.676	0.292	0.000	0.000	0.000	2.325E-01	-1.509E-01	-2.357E-01	1.527E-01	3.046E-04
6.00	1.371	-0.944	0.292	0.000	0.000	0.000	2.299E-01	-1.282E-01	-2.446E-01	1.545E-01	-6.888E-04
8.00	1.825	-1.188	0.290	0.000	0.000	0.000	2.157E-01	-1.153E-01	-2.486E-01	1.534E-01	-1.725E-03
10.00	2.234	-1.405	0.289	0.000	0.000	0.000	1.978E-01	-4.823E-02	-2.463E-01	1.536E-01	-3.307E-03
12.00	2.616	-1.382	0.287	0.000	0.000	0.000	2.113E-01	-5.773E-02	-2.611E-01	2.223E-01	-8.383E-03
14.00	3.080	-1.637	0.285	0.000	0.000	0.000	2.420E-01	-1.336E-01	-2.776E-01	2.943E-01	-1.433E-02
16.00	3.584	-1.916	0.282	0.000	0.000	0.000	2.725E-01	-1.659E-01	-2.882E-01	3.347E-01	-2.068E-02
18.00	4.170	-2.301	0.278	0.000	0.000	0.000	2.893E-01	-1.648E-01	-2.985E-01	3.728E-01	-2.779E-02
20.00	4.742	-2.576	0.272	0.000	0.000	0.000	2.826E-01	-1.104E-01	-3.092E-01	4.162E-01	-3.487E-02

ALPHA	CL	CD	CL/CD	X-C.P.
0.00	0.000	0.293	0.000	-0.796
2.00	0.431	0.308	1.399	-0.770
4.00	0.883	0.355	2.489	-0.746
6.00	1.333	0.433	3.077	-0.688
8.00	1.767	0.542	3.262	-0.651
10.00	2.150	0.673	3.196	-0.629
12.00	2.500	0.825	3.029	-0.528
14.00	2.919	1.022	2.857	-0.532
16.00	3.368	1.259	2.675	-0.535
18.00	3.880	1.553	2.499	-0.552
20.00	4.363	1.877	2.324	-0.543

PANEL DEFLECTION ANGLES (DEGREES)

FIN SET	FIN 1	FIN 2	FIN 3	FIN 4
1	0.00	0.00	0.00	0.00
2	0.00	0.00	0.00	0.00

THE USAF AUTOMATED MISSILE DATCOM * REV 5/93 *
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
ROCKPRN
BODY + 2 FIN SETS DYNAMIC DERIVATIVES

FLIGHT CONDITIONS								REFERENCE DIMENSIONS				
MACH NUMBER	ALTITUDE	VELOCITY	PRESSURE	TEMPERATURE	REYNOLDS NUMBER	SIDESLIP ANGLE	ROLL ANGLE	REF. AREA	REF. LENGTH LONG.	REF. LENGTH LAT.	MOMENT REF. CENTER LONG.	MOMENT REF. CENTER VERTICAL
	FT	FT/SEC	LB/FT**2	DEG R	1/FT	DEG	DEG	FT**2	FT	FT	FT	FT
0.85	0.00	948.83	2.116E+03	518.67	6.008E+06	0.00	0.00	1.230	1.250	1.250	5.458	0.000

ALPHA	DYNAMIC DERIVATIVES (PER DEGREE) CNQ	CNAD	CMQ+CMAD
0.0	1.809E+00	5.125E-01	-4.66958E+00
2.0	1.963E+00	5.127E-01	-5.00010E+00
4.0	2.119E+00	5.302E-01	-5.34752E+00
6.0	2.293E+00	6.757E-01	-5.83525E+00
8.0	2.458E+00	7.839E-01	-6.27272E+00
10.0	2.641E+00	1.023E+00	-6.85188E+00
12.0	2.845E+00	1.413E+00	-7.59618E+00
14.0	3.023E+00	1.681E+00	-8.18907E+00
16.0	3.204E+00	1.991E+00	-8.81956E+00
18.0	3.376E+00	2.279E+00	-9.41075E+00
20.0	3.534E+00	2.504E+00	-9.92414E+00

APPENDIX C - CTA.DAT

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64       - # OF DATA SETS - NDS
0.3751   - DISTANCE FROM CENTERLINE TO CENTROID OF FIN (lref units) - YCENT
2        - NUMBER OF FIN SETS - NFS
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APPENDIX D - CART.OUT

```
 8.000000
 11.00000

0.20      0.00                                        0.141E+07   0.00    0.00    1.230     1.250         5.458     0.000

  0.00    0.000   0.000   0.193     0.000   0.000   0.000   0.198E+01  -0.220E+01  -0.206E+01   0.221E+01  -0.698E+00
  2.00    0.423  -0.445   0.193     0.000   0.000   0.000   0.225E+01  -0.225E+01  -0.218E+01   0.219E+01   0.147E+01
  4.00    0.899  -0.900   0.193     0.000   0.000   0.000   0.237E+01  -0.175E+01  -0.239E+01   0.214E+01   0.699E+01
  6.00    1.372  -1.146   0.192     0.000   0.000   0.000   0.235E+01  -0.139E+01  -0.261E+01   0.244E+01  -0.216E+01
  8.00    1.838  -1.456   0.191     0.000   0.000   0.000   0.223E+01  -0.144E+01  -0.270E+01   0.243E+01  -0.269E+01
 10.00    2.264  -1.721   0.190     0.000   0.000   0.000   0.224E+01  -0.178E+01  -0.271E+01   0.244E+01  -0.364E+01
 12.00    2.734  -2.169   0.188     0.000   0.000   0.000   0.232E+01  -0.192E+01  -0.276E+01   0.262E+01  -0.408E+01
 14.00    3.193  -2.491   0.185     0.000   0.000   0.000   0.230E+01  -0.155E+01  -0.287E+01   0.313E+01  -0.802E+01
 16.00    3.654  -2.788   0.182     0.000   0.000   0.000   0.236E+01  -0.156E+01  -0.293E+01   0.348E+01  -0.122E+01
 18.00    4.137  -3.114   0.179     0.000   0.000   0.000   0.244E+01  -0.140E+01  -0.293E+01   0.380E+01  -0.181E+01
 20.00    4.631  -3.348   0.175     0.000   0.000   0.000   0.249E+01  -0.940E+01  -0.297E+01   0.417E+01  -0.232E+01
       -1.111
       -1.053
       -1.001
       -0.835
       -0.792
       -0.760
       -0.793
       -0.780
       -0.763
       -0.753
       -0.723
  2       0.00        0.00        0.00       0.00
      0.160E+01           0.604E+00         -0.44443E+01
      0.176E+01           0.987E+00         -0.47407E+01
      0.191E+01           0.134E+01         -0.50265E+01
      0.203E+01           0.151E+01         -0.52556E+01
      0.213E+01           0.154E+01         -0.54434E+01
      0.223E+01           0.159E+01         -0.56268E+01
      0.217E+01           0.157E+01         -0.53051E+01
      0.194E+01           0.118E+01         -0.44626E+01
      0.164E+01           0.371E+00         -0.35447E+01
      0.142E+01           0.171E+00         -0.27057E+01
      0.129E+01           0.685E+00         -0.19965E+01
0.60      0.00                                        0.424E+07   0.00    0.00    1.230     1.250         5.458     0.000

  0.00    0.000   0.000   0.192     0.000   0.000   0.000   0.206E+01  -0.230E+01  -0.212E+01   0.230E+01  -0.303E+01
  2.00    0.431  -0.460   0.192     0.000   0.000   0.000   0.225E+01  -0.230E+01  -0.221E+01   0.224E+01   0.124E+01
  4.00    0.901  -0.918   0.191     0.000   0.000   0.000   0.232E+01  -0.206E+01  -0.239E+01   0.215E+01   0.570E+01
  6.00    1.359  -1.285   0.191     0.000   0.000   0.000   0.229E+01  -0.186E+01  -0.249E+01   0.216E+01  -0.197E+01
  8.00    1.819  -1.661   0.190     0.000   0.000   0.000   0.221E+01  -0.170E+01  -0.254E+01   0.210E+01  -0.429E+01
 10.00    2.243  -1.964   0.189     0.000   0.000   0.000   0.205E+01  -0.103E+01  -0.255E+01   0.212E+01  -0.181E+01
 12.00    2.638  -2.074   0.188     0.000   0.000   0.000   0.210E+01  -0.106E+01  -0.268E+01   0.271E+01  -0.593E+01
 14.00    3.082  -2.389   0.186     0.000   0.000   0.000   0.241E+01  -0.167E+01  -0.281E+01   0.332E+01  -0.110E+01
 16.00    3.602  -2.743   0.183     0.000   0.000   0.000   0.280E+01  -0.199E+01  -0.291E+01   0.367E+01  -0.166E+01
 18.00    4.203  -3.185   0.179     0.000   0.000   0.000   0.298E+01  -0.193E+01  -0.300E+01   0.399E+01  -0.235E+01
 20.00    4.796  -3.518   0.174     0.000   0.000   0.000   0.295E+01  -0.139E+01  -0.312E+01   0.445E+01  -0.310E+01
       -1.121
       -1.068
       -1.019
       -0.945
       -0.913
       -0.876
       -0.786
```

```
         -0.775
         -0.762
         -0.758
         -0.733
  2        0.00      0.00      0.00      0.00
        0.174E+01           0.584E+00        -0.53456E+01
        0.195E+01           0.728E+00        -0.59245E+01
        0.212E+01           0.659E+00        -0.64218E+01
        0.230E+01           0.628E+00        -0.69254E+01
        0.252E+01           0.862E+00        -0.75174E+01
        0.273E+01           0.105E+01        -0.80836E+01
        0.289E+01           0.981E+00        -0.85395E+01
        0.309E+01           0.121E+01        -0.90932E+01
        0.334E+01           0.169E+01        -0.97267E+01
        0.353E+01           0.192E+01        -0.10249E+02
        0.372E+01           0.215E+01        -0.10763E+02

 0.85      0.00                                  0.601E+07     0.00     0.00     1.230     1.250              5.458       0.000

  0.00     0.000    0.000    0.293     0.000    0.000    0.000      0.215E+01  -0.171E+01  -0.219E+01   0.171E+01   0.442E+01
  2.00     0.441   -0.340    0.293     0.000    0.000    0.000      0.226E+01  -0.169E+01  -0.225E+01   0.164E+01   0.729E+01
  4.00     0.906   -0.676    0.292     0.000    0.000    0.000      0.233E+01  -0.151E+01  -0.236E+01   0.153E+01   0.305E+01
  6.00     1.371   -0.944    0.292     0.000    0.000    0.000      0.230E+01  -0.128E+01  -0.245E+01   0.154E+01  -0.689E+01
  8.00     1.825   -1.188    0.290     0.000    0.000    0.000      0.216E+01  -0.115E+01  -0.249E+01   0.153E+01  -0.173E+01
 10.00     2.234   -1.405    0.289     0.000    0.000    0.000      0.198E+01  -0.482E+01  -0.246E+01   0.154E+01  -0.331E+01
 12.00     2.616   -1.382    0.287     0.000    0.000    0.000      0.211E+01  -0.577E+01  -0.261E+01   0.222E+01  -0.838E+01
 14.00     3.080   -1.637    0.285     0.000    0.000    0.000      0.242E+01  -0.134E+01  -0.278E+01   0.294E+01  -0.143E+01
 16.00     3.584   -1.916    0.282     0.000    0.000    0.000      0.272E+01  -0.166E+01  -0.288E+01   0.335E+01  -0.207E+01
 18.00     4.170   -2.301    0.278     0.000    0.000    0.000      0.289E+01  -0.165E+01  -0.298E+01   0.373E+01  -0.278E+01
 20.00     4.742   -2.576    0.272     0.000    0.000    0.000      0.283E+01  -0.110E+01  -0.309E+01   0.416E+01  -0.349E+01
         -0.796
         -0.770
         -0.746
         -0.688
         -0.651
         -0.629
         -0.528
         -0.532
         -0.535
         -0.552
         -0.543
  2        0.00      0.00      0.00      0.00
        0.181E+01           0.512E+00        -0.46696E+01
        0.196E+01           0.513E+00        -0.50001E+01
        0.212E+01           0.530E+00        -0.53475E+01
        0.229E+01           0.676E+00        -0.58352E+01
        0.246E+01           0.784E+00        -0.62727E+01
        0.264E+01           0.102E+01        -0.68519E+01
        0.285E+01           0.141E+01        -0.75962E+01
        0.302E+01           0.168E+01        -0.81891E+01
        0.320E+01           0.199E+01        -0.88196E+01
        0.338E+01           0.228E+01        -0.94108E+01
        0.353E+01           0.250E+01        -0.99241E+01

 0.95      0.00                                  0.672E+07     0.00     0.00     1.230     1.250              5.458       0.000

  0.00     0.000    0.000    0.470     0.000    0.000    0.000      0.226E+01  -0.179E+01  -0.230E+01   0.177E+01  -0.861E+01
  2.00     0.461   -0.351    0.470     0.000    0.000    0.000      0.235E+01  -0.173E+01  -0.234E+01   0.167E+01   0.539E+01
  4.00     0.939   -0.690    0.469     0.000    0.000    0.000      0.242E+01  -0.151E+01  -0.243E+01   0.151E+01   0.219E+01
  6.00     1.429   -0.955    0.468     0.000    0.000    0.000      0.239E+01  -0.134E+01  -0.254E+01   0.152E+01  -0.107E+01
```

APPENDIX E - AEROB.OUT

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C  NALPHA= 11  NMACH=  8
C AOA ACROSS (DEG): 0.0, 2.0, 4.0, 6.0, 8.0,10.0,12.0,14.0,16.0,18.0,20.0,
     DATA ALPHA/ 0.0, 2.0, 4.0, 6.0, 8.0,10.0,12.0,14.0,16.0,18.0,20.0,
C M# DOWN:0.20,0.60,0.85,0.95,1.00,1.10,1.60,2.00,
     DATA MACH/0.20,0.60,0.85,0.95,1.00,1.10,1.60,2.00,
      DATA TCN/
    &  0.000,  0.423,  0.899,  1.372,  1.838,  2.264,  2.734,  3.193,  3.654,  4.137,  4.631,
    &  0.000,  0.431,  0.901,  1.359,  1.819,  2.243,  2.638,  3.082,  3.602,  4.203,  4.796,
    &  0.000,  0.441,  0.906,  1.371,  1.825,  2.234,  2.616,  3.080,  3.584,  4.170,  4.742,
    &  0.000,  0.461,  0.939,  1.429,  1.897,  2.329,  2.734,  3.222,  3.756,  4.369,  4.945,
    &  0.000,  0.481,  0.977,  1.484,  1.969,  2.419,  2.843,  3.357,  3.922,  4.560,  5.154,
    &  0.000,  0.458,  0.929,  1.409,  1.869,  2.297,  2.697,  3.191,  3.719,  4.295,  4.876,
    &  0.000,  0.429,  0.881,  1.342,  1.792,  2.207,  2.586,  3.033,  3.893,  4.679,  5.423,
    &  0.000,  0.396,  0.814,  1.251,  1.672,  2.054,  2.596,  3.190,  3.821,  4.502,  5.131,
      DATA TCNP/
    &  0.000,  0.004,  0.022,  0.065,  0.099,  0.163,  0.222,  0.293,  0.371,  0.461,  0.570,
    &  0.000,  0.005,  0.021,  0.048,  0.065,  0.124,  0.234,  0.316,  0.420,  0.556,  0.739,
    &  0.000,  0.004,  0.013,  0.032,  0.045,  0.098,  0.215,  0.304,  0.415,  0.559,  0.754,
    &  0.000,  0.003,  0.013,  0.029,  0.035,  0.101,  0.230,  0.329,  0.454,  0.616,  0.835,
    &  0.000,  0.003,  0.012,  0.025,  0.028,  0.097,  0.240,  0.351,  0.489,  0.668,  0.912,
    &  0.000,  0.003,  0.009,  0.021,  0.022,  0.104,  0.240,  0.346,  0.484,  0.660,  0.910,
    &  0.000,  0.002,  0.010,  0.021,  0.034,  0.073,  0.141,  0.190,  0.282,  0.380,  0.501,
    &  0.000,  0.001,  0.009,  0.020,  0.034,  0.064,  0.145,  0.203,  0.270,  0.352,  0.451,
      DATA TCLM/
    &  0.000, -0.445, -0.900, -1.146, -1.456, -1.721, -2.169, -2.491, -2.788, -3.114, -3.348,
    &  0.000, -0.460, -0.918, -1.285, -1.661, -1.964, -2.074, -2.389, -2.743, -3.185, -3.518,
    &  0.000, -0.340, -0.676, -0.944, -1.188, -1.405, -1.382, -1.637, -1.916, -2.301, -2.576,
    &  0.000, -0.351, -0.690, -0.955, -1.226, -1.392, -1.375, -1.642, -1.936, -2.326, -2.562,
    &  0.000, -0.442, -0.866, -1.205, -1.547, -1.780, -1.830, -2.178, -2.564, -3.041, -3.332,
    &  0.000, -0.611, -1.211, -1.730, -2.244, -2.650, -2.846, -3.355, -3.859, -4.428, -4.819,
    &  0.000, -0.451, -0.922, -1.364, -1.764, -2.094, -2.232, -2.573, -3.164, -3.756, -4.228,
    &  0.000, -0.397, -0.816, -1.218, -1.578, -1.874, -2.028, -2.404, -2.823, -3.321, -3.712,
      DATA TCLMP/
    &  0.000,  0.004,  0.022, -0.059, -0.061, -0.222, -0.384, -0.619, -0.937, -1.342, -1.855,
    &  0.000,  0.006,  0.028,  0.038,  0.097, -0.068, -0.490, -0.810, -1.231, -1.792, -2.583,
    &  0.000,  0.006,  0.033,  0.049,  0.095, -0.076, -0.557, -0.903, -1.358, -1.957, -2.792,
    &  0.000,  0.007,  0.041,  0.064,  0.147, -0.077, -0.606, -0.998, -1.509, -2.191, -3.151,
    &  0.000,  0.009,  0.049,  0.079,  0.171, -0.063, -0.653, -1.102, -1.670, -2.428, -3.507,
    &  0.000,  0.009,  0.044,  0.067,  0.144, -0.181, -0.773, -1.217, -1.813, -2.611, -3.738,
    &  0.000,  0.000,  0.003, -0.008, -0.014, -0.155, -0.437, -0.620, -0.871, -1.174, -1.601,
    &  0.000, -0.002, -0.003, -0.018, -0.033, -0.133, -0.414, -0.584, -0.785, -1.031, -1.371,
      DATA TCYP/
    &  0.000, -0.002, -0.015, -0.064, -0.112, -0.170, -0.217, -0.310, -0.402, -0.480, -0.518,
    &  0.000, -0.003, -0.015, -0.042, -0.069, -0.111, -0.210, -0.309, -0.432, -0.565, -0.695,
    &  0.000, -0.001, -0.011, -0.032, -0.057, -0.089, -0.196, -0.303, -0.430, -0.566, -0.695,
    &  0.000, -0.002, -0.010, -0.031, -0.050, -0.096, -0.210, -0.332, -0.476, -0.628, -0.771,
    &  0.000, -0.002, -0.009, -0.030, -0.049, -0.098, -0.221, -0.356, -0.516, -0.683, -0.845,
    &  0.000, -0.001, -0.007, -0.027, -0.047, -0.099, -0.225, -0.356, -0.509, -0.667, -0.823,
    &  0.000, -0.001, -0.007, -0.020, -0.039, -0.070, -0.147, -0.215, -0.339, -0.452, -0.564,
    &  0.000, -0.001, -0.007, -0.017, -0.035, -0.057, -0.147, -0.222, -0.302, -0.386, -0.475,
      DATA TCLNP/
    &  0.000, -0.002, -0.011,  0.130,  0.199,  0.311,  0.381,  0.680,  1.034,  1.388,  1.579,
    &  0.000, -0.003, -0.015,  0.018,  0.025,  0.106,  0.454,  0.822,  1.284,  1.798,  2.328,
    &  0.000, -0.003, -0.017,  0.009,  0.034,  0.102,  0.500,  0.892,  1.376,  1.896,  2.416,
    &  0.000, -0.004, -0.022,  0.005,  0.003,  0.119,  0.547,  0.993,  1.542,  2.131,  2.728,
    &  0.000, -0.005, -0.027,  0.001,  0.000,  0.125,  0.595,  1.097,  1.717,  2.382,  3.062,
    &  0.000, -0.004, -0.023,  0.012,  0.029,  0.195,  0.709,  1.226,  1.850,  2.526,  3.181,
    &  0.000,  0.001,  0.000,  0.029,  0.073,  0.161,  0.465,  0.714,  1.025,  1.333,  1.648,
    &  0.000,  0.001,  0.003,  0.027,  0.063,  0.128,  0.455,  0.681,  0.935,  1.209,  1.534,
      DATA TCA/
```

```
&  0.193,  0.192,  0.293,  0.470,  0.570,  0.642,  0.606,  0.553,
  DATA TCAA/
&-0.00090,-0.00090,-0.00105,-0.00145,-0.00175,-0.00205,-0.00180,-0.00235,
  DATA TCAD/
& 0.00576, 0.00688, 0.00784, 0.00880, 0.00976, 0.01040, 0.00464, 0.00352,
  DATA TCNDQ/
& 0.07840, 0.09620, 0.10820, 0.12180, 0.13720, 0.13700, 0.05380, 0.03880,
  DATA TCLMDQ/
&-0.37620,-0.46100,-0.51860,-0.58280,-0.65360,-0.67160,-0.26480,-0.19020,
  DATA TCLMQ/
&  -4.4443,  -5.3456,  -4.6696,  -3.6102,  -3.2275,  -2.1108,  -0.5894,  -0.5673,
  DATA TCLLAP/
&-0.00016,-0.00018,-0.00027,-0.00032,-0.00037,-0.00042,-0.00023,-0.00024,
  DATA TCLLDP/
& 0.01900, 0.02260, 0.02560, 0.02880, 0.03240, 0.03480, 0.01560, 0.01160,
  DATA TCLLP/
&-0.01226,-0.01458,-0.01652,-0.01858,-0.02090,-0.02245,-0.01006,-0.00748,
```

APPENDIX F - CTA.FOR

```
C CTA4.FOR  29 NOV 94
C THIS PROGRAM CONVERTS MISSILE DATCOM OUTPUT IN CARTESIAN COORDINATES
C TO AEROBALLISTIC COORDINATES.  GREG WILDER
C                                 ASC/XREWS  904-882-3722
C2345678901234567892123456789312345678941234567895123456789612345678971 2
C
      DIMENSION ALPHA(20),BC(16,20,100),CN(16,20),CNP(16,20),
     & CLM(16,20),CLMP(16,20),CYP(16,20),CLNP(16,20),CA(16),
     & CAA(16),CAD(16),CNDQ(16),CLMDQ(16),CLLDP(16),CLLP(16),
     & CLMQ(16),CLLAP(16)
      REAL LREF
      CHARACTER*6 ANUM,ADYN,ACASE
C
C OPEN INPUT FILE & OUTPUT FILE
      OPEN(UNIT=5,FILE='CTA.DAT',STATUS='OLD')
      OPEN(UNIT=6,FILE='FOR006.DAT',STATUS='OLD')
      OPEN(UNIT=9,FILE='CART.OUT',STATUS='NEW')
      OPEN(UNIT=10,FILE='AEROB.OUT',STATUS='NEW')
C
C VARIABLES:
C ALPHA - ALPHA MATRIX
C BC - BODY COEFFICEINTS FROM DATCOM - MAX 20 ALPHA, MAX 16 M#
C      & 100 DATA SETS
C NDS - # OF DATA SETS
C YCENT - DISTANCE FROM BODY CENTERLINE TO CENTROID OF CONTROL FIN
C         SAME UNITS AS LREF
C NFIN  - NUMBER OF THE FIN SET TO BE DEFLECTED
C RAD - DEGREES TO RADIANS CONVERSION FACTOR
C DEG - RAD TO DEG CONVERSION FACTOR
C I22 - FACTOR REQUIRED TO SKIP TO RUN WITH EQUAL M# @22.5 DEG
C I45 - FACTOR REQUIRED TO SKIP TO RUN WITH EQUAL M# @45 DEG
C
      RAD = 1.0/57.2958
      DEG = 57.2958
      READ(5,*)NDS
      READ(5,*)YCENT
      READ(5,*)NFIN
C
C READ IN # OF M# & # OF AOA'S (LINE 16 & 17) FROM INPUT DECK ECHO
C
      DO 100 ILOOK=1,999
      READ(6,1000)ACASE
1000  FORMAT(5X,A6)
 100  IF (ACASE .EQ. 'CASEID')GOTO 10
C
C READ IN NMACH & NALP (REAL IN BC)
C
10    READ(6,1010)BC(2,1,1)
      WRITE(9,*)BC(2,1,1)
1010  FORMAT(21X,F2.0)
      READ(6,1015)BC(1,1,1)
      WRITE(9,*)BC(1,1,1)
1015  FORMAT(23X,F2.0)
C
C BEGIN READ OF DATA
```

```
C
      DO 110 I=1,NDS
C
C # OF M# & AOA'S ARE CONSTANT
      BC(1,1,I)=BC(1,1,1)
      BC(2,1,I)=BC(2,1,1)
C
C LOCATE FIRST/NEXT DATA SET BY FINDING "NUMBER" (IN MACH NUMBER)
C
      DO 120 ILOOK=1,10000
15     READ(6,1020)ANUM
1020   FORMAT(1X,A6)
       IF(ANUM .EQ. 'NUMBER')GOTO 20
120   CONTINUE
C
C READ HEADER INFO AND INSERT INTO FIRST COL OF BC MATRIX
C
20    READ(6,1040)BC(3,1,I),BC(4,1,I),BC(5,1,I),BC(6,1,I),BC(7,1,I),
     & BC(8,1,I),BC(9,1,I),BC(10,1,I),BC(11,1,I)
C
      WRITE(9,1041)BC(3,1,I),BC(4,1,I),BC(5,1,I),BC(6,1,I),BC(7,1,I),
     & BC(8,1,I),BC(9,1,I),BC(10,1,I),BC(11,1,I)
C
1040  FORMAT(/,2X,F4.2,2X,F10.2,37X,E9.3,2X,F8.2,2X,F7.2,2X,F8.3,2X,
     & F7.3,11X,F8.3,2X,F9.3,//////)
1041  FORMAT(/,2X,F4.2,2X,F10.2,37X,E9.3,2X,F8.2,2X,F7.2,2X,F8.3,2X,
     & F7.3,11X,F8.3,2X,F9.3,/)
C
C READ FIRST BLOCK OF DATA
C23456789012345678921234567893123456789412345678951234567896123456789712
      NALPHA=IFIX ( BC(1,1,1) )
      DO 125 J=1,NALPHA
       READ(6,1050)ALPHA(J),BC(J,2,I),BC(J,3,I),BC(J,4,I),BC(J,5,I),
     & BC(J,6,I),BC(J,7,I),BC(J,8,I),BC(J,9,I),BC(J,10,I),BC(J,11,I),
     & BC(J,12,I)
       WRITE(9,1050)ALPHA(J),BC(J,2,I),BC(J,3,I),BC(J,4,I),
     & BC(J,5,I),BC(J,6,I),BC(J,7,I),BC(J,8,I),BC(J,9,I),
     & BC(J,10,I),BC(J,11,I),BC(J,12,I)
1050   FORMAT(3X,F5.2,1X,3(2X,F7.3),4X,3(2X,F7.3),4X,5(2X,E10.3))
C
125    CONTINUE
C READ IN XCP
      READ(6,1055)
1055  FORMAT(//)
      DO 130 J=1,NALPHA
      READ(6,1060)BC(J,13,I)
      WRITE(9,1061)BC(J,13,I)
1060  FORMAT(78X,F8.3)
1061  FORMAT(8X,F8.3)
C
130    CONTINUE
C
C READ IN FIN DEFLECTIONS (IF NFIN = 2 SKIP TO SECOND FIN SET)
C
      READ(6,1065)
1065  FORMAT(/)
C      IF(NFIN .EQ. 2)READ(6,1065)
132   READ(6,1066)IFS,BC(12,1,I),BC(13,1,I),BC(14,1,I),BC(15,1,I)
      IF (IFS .LT. NFIN)GOTO 132
      WRITE(9,1067)IFS,BC(12,1,I),BC(13,1,I),BC(14,1,I),BC(15,1,I)
```

```
1066  FORMAT(43X,I2,4(4X,F7.2))
1067  FORMAT(3X,I2,4(4X,F7.2))
C
C READ IN DYNAMIC DERIV
C -LOCATE DYNAMIC DERIV DATA BY FINDING "DYNAMI" (IN HEADER)
C
      DO 140 ILOOK=1,10000
      READ(6,1070)ADYN
1070  FORMAT(49X,A6)
      IF(ADYN .EQ. 'DYNAMI')GOTO 30
140    CONTINUE
C
30    READ(6,1075)
1075  FORMAT(/)
      DO 150 J=1,NALPHA
       READ(6,1080)BC(J,14,I),BC(J,15,I),BC(J,16,I)
       WRITE(9,1085)BC(J,14,I),BC(J,15,I),BC(J,16,I)
1080   FORMAT(47X,E11.3,10X,E10.3,11X,E13.5)
1085   FORMAT(8X,E10.3,10X,E10.3,11X,E13.5)
150   CONTINUE
C
110   CONTINUE
C
C CALCULATE CN - AEROBALLISTIC CN PRIME
C            CNP - CORRECTION TO CN WHEN PHIP IS NOT ZERO. DELTA CNP
C                  = CN @ PHIP = 45 - CN @ PHIP = 0
C TCNP45 - TEMP CN @ PHIP = 45 DEG
C            CLM - AEROBALLISTIC CM - @ PHIP = 0
C            CLMP - CORR TO CLM WHEN PHIP IS NOT ZERO.
C                   DELTA CLM = CLM @ PHIP = 45 - CLM @ PHIP = 0
C            CA - CA - AXIAL COEF AT ZERO AOA
C
      NMACH=IFIX(BC(2,1,1))
      WRITE(9,1100)NMACH,BC(2,1,1)
1100  FORMAT(2X,'NMACH = ',I4,'BC(2,1,1) = ',F10.4)
      I22 = NMACH
      I45 = 2*NMACH
C
      DO 200 I=1,NMACH
        DO 210 J=1,NALPHA
C
          CN(I,J) = BC(J,2,I)
C
          PHIP = RAD * BC(7,1,I+I45)
          TCNP45 = BC(J,2,I+I45) * COS(PHIP) - BC(J,5,I+I45) * SIN(PHIP)
          CNP(I,J) = TCNP45 - BC(J,2,I)
C
          CLM(I,J) = BC(J,3,I)
          TCMP45 = BC(J,3,I+I45) * COS(PHIP) - BC(J,6,I+I45) * SIN(PHIP)
          CLMP(I,J) = TCMP45 - BC(J,3,I)
C
          PHIP = RAD * BC(7,1,I+I22)
C
          CYP(I,J) = BC(J,5,I+I22) * COS(PHIP) +
     &                         BC(J,2,I+I22) * SIN(PHIP)
          CLNP(I,J) = BC(J,3,I+I22)*SIN(PHIP)+BC(J,6,I+I22)*COS(PHIP)
C
210     CONTINUE
C
      CA(I) = BC(1,4,I)
```

```
C
200   CONTINUE
C
C CALC CAA & CAD
C
      DO 300 I=1,NMACH
C
C CAA
C
        CASUM = 0.0
        DO 310 J=1,NALPHA-1
          DALP= ALPHA(J+1)-ALPHA(J)
          CASUM = (BC(J+1,4,I) - BC(J,4,I))/DALP + CASUM
310     CONTINUE
C
       CAA(I) = CASUM / (NALPHA-1)
C
C CALC DQ, DR & DEFF
C
        D1 = BC(12,1,I+6*NMACH)
        D2 = BC(13,1,I+6*NMACH)
        D3 = BC(14,1,I+6*NMACH)
        D4 = BC(15,1,I+6*NMACH)
        DQ = (-D1-D2+D3+D4)/4.0
        DR = (-D1+D2+D3-D4)/4.0
        DEFF = ( ABS(DQ) + ABS(DR) ) / 2.0
C
C CAD (CALC @ ZERO AOA ONLY.)
C
        J = 1
        CAESUM = (BC(J,4,I+6*NMACH) - BC(J,4,I))
C
       CAD(I) = CAESUM / (DEFF**2)
C
C CNDQ & CLMDQ LOOP @ ZERO AOA
C
C CALC DQ
C
        D1 = BC(12,1,I+4*NMACH)
        D2 = BC(13,1,I+4*NMACH)
        D3 = BC(14,1,I+4*NMACH)
        D4 = BC(15,1,I+4*NMACH)
        DQ = (-D1-D2+D3+D4)/4.0
C
        J=1
        CNDQSUM = (BC(J,2,I) - BC(J,2,I+4*NMACH))
        CLMDQSUM = (BC(J,3,I) - BC(J,3,I+4*NMACH))
C
       CNDQ(I) = CNDQSUM/(DQ)
       CLMDQ(I) = CLMDQSUM/(DQ)
C
C CLLAP
C
        CLSUM = 0.0
        NSUM = 0
        DO 325 J=1,NALPHA
          ACLL = ABS(BC(J,7,I+NMACH))
          IF ( ACLL .LT. 0.001 ) GO TO 325
C ONLY NON-ZERO CLL VALUES ARE INCLUDED IN THIS CALCULATION
          CLSUM = ((BC(J,7,I+NMACH)-BC(J,7,I))/ALPHA(J)**2) + CLSUM
```

```
            NSUM = NSUM + 1
 325      CONTINUE
          CLLAP(I) = CLSUM/NSUM
C
C CLLDP @ ZERO AOA
C
C CALC DP
C
          D1 = BC(12,1,I+3*NMACH)
          D2 = BC(13,1,I+3*NMACH)
          D3 = BC(14,1,I+3*NMACH)
          D4 = BC(15,1,I+3*NMACH)
          DP = (-D1-D2-D3-D4)/4.0
C
          J=1
          CLLDP(I) = (BC(J,7,I+3*NMACH) - BC(J,7,I))/(DP)
C
C CALC CLLP
C
         LREF = BC(9,1,1)
         CLLP(I) = -2.15*(YCENT/LREF)*CLLDP(I)
C
C CALC CLMQ
C
         J=1
         CLMQ(I) = BC(J,16,I)
C
300    CONTINUE
C
C PRINT COEFFICIENTS
C
C WRITE ALPHA & M# INFORMATION
C
          WRITE(10,1200)NALPHA,NMACH
1200      FORMAT(1X,'C  NALPHA=',I3,2X,'NMACH=',I3)
          WRITE(10,1210)(ALPHA(J),J=1,NALPHA)
1210      FORMAT(1X,'C AOA ACROSS (DEG):',20(F4.1,','))
          WRITE(10,1212)(ALPHA(J),J=1,NALPHA)
1212      FORMAT(6X,'DATA ALPHA/',20(F4.1,','))
          WRITE(10,1220)(BC(3,1,J),J=1,NMACH)
1220      FORMAT(1X,'C M# DOWN:',16(F4.2,','))
          WRITE(10,1222)(BC(3,1,J),J=1,NMACH)
1222      FORMAT(6X,'DATA MACH/',16(F4.2,','))
C
C WRITE CN
C
          WRITE(10,1230)
1230      FORMAT(7X,'DATA TCN/')
C
          DO 320 I=1,NMACH
320         WRITE(10,1250)(CN(I,J),J=1,NALPHA)
C
C WRITE CNP
C
          WRITE(10,1240)
1240      FORMAT(7X,'DATA TCNP/')
          DO 340 I=1,NMACH
340         WRITE(10,1250)(CNP(I,J),J=1,NALPHA)
C
C WRITE CLM
```

```
C
         WRITE(10,1245)
1245     FORMAT(7X,'DATA TCLM/')
         DO 345 I=1,NMACH
345        WRITE(10,1250)(CLM(I,J),J=1,NALPHA)
C
C WRITE CLMP
C
         WRITE(10,1260)
1260     FORMAT(7X,'DATA TCLMP/')
         DO 350 I=1,NMACH
350        WRITE(10,1250)(CLMP(I,J),J=1,NALPHA)
C
C WRITE CYP
C
         WRITE(10,1270)
1270     FORMAT(7X,'DATA TCYP/')
         DO 360 I=1,NMACH
360        WRITE(10,1250)(CYP(I,J),J=1,NALPHA)
C
C WRITE CLNP
C
         WRITE(10,1280)
1280     FORMAT(7X,'DATA TCLNP/')
         DO 370 I=1,NMACH
370        WRITE(10,1250)(CLNP(I,J),J=1,NALPHA)
C
C WRITE CA
C
         WRITE(10,1290)
1290     FORMAT(7X,'DATA TCA/')
C
         WRITE(10,1250)(CA(I),I=1,NMACH)
C
C WRITE CAA
C
         WRITE(10,1300)
1300     FORMAT(7X,'DATA TCAA/')
         WRITE(10,1310)(CAA(I),I=1,NMACH)
1310     FORMAT(5X,'&',20(F8.5,','))
1312     FORMAT(5X,'&',20(F9.4,','))
C
1250     FORMAT(5X,'&',20(F7.3,','))
C
C WRITE CAD
C
         WRITE(10,1320)
1320     FORMAT(7X,'DATA TCAD/')
         WRITE(10,1310)(CAD(I),I=1,NMACH)
C
C WRITE CNDQ
C
         WRITE(10,1330)
1330     FORMAT(7X,'DATA TCNDQ/')
         WRITE(10,1310)(CNDQ(I),I=1,NMACH)
C
C WRITE CLMDQ
C
         WRITE(10,1340)
1340     FORMAT(7X,'DATA TCLMDQ/')
```

```
      WRITE(10,1310)(CLMDQ(I),I=1,NMACH)
C
C WRITE CLMQ
C
      WRITE(10,1370)
1370  FORMAT(7X,'DATA TCLMQ/')
      WRITE(10,1312)(CLMQ(I),I=1,NMACH)
C
C
C WRITE CLLAP
C
      WRITE(10,1345)
1345  FORMAT(7X,'DATA TCLLAP/')
      WRITE(10,1310)(CLLAP(I),I=1,NMACH)
C
C WRITE CLLDP
C
      WRITE(10,1350)
1350  FORMAT(7X,'DATA TCLLDP/')
      WRITE(10,1310)(CLLDP(I),I=1,NMACH)
C
C WRITE CLLP
C
      WRITE(10,1360)
1360  FORMAT(7X,'DATA TCLLP/')
      WRITE(10,1310)(CLLP(I),I=1,NMACH)
C
      STOP
      END
```

APPENDIX G - CTA EQUATION SUMMARY

$$CA = C_{Ao} = C_{A(\alpha'=0°)} = f(M^{\#})$$

$$CAA = C_{A_{\alpha'}} = \frac{\partial C_A}{\partial \alpha'} \approx \frac{\sum_{I=1}^{\#\alpha-1} \frac{\Delta C_A}{\Delta \alpha}}{(\#\alpha - 1)} = f(M^{\#})$$

$$CAD = C_{A_{\delta^2 eff}} = \frac{\partial C_A}{\partial \delta_{eff}^2} \approx \frac{\Delta C_A}{\Delta \delta_{eff}^2} = \frac{C_{A(\delta_{eff}=5°,\alpha'=0°)} - C_{A(\delta_{eff}=0°,\alpha'=0°)}}{\delta_{eff}^2} = f(M^{\#})$$

$$CYP = \Delta C'_{Y,\phi'} = C_{Y_{(\phi'=22.5°)}} \cos(\phi') + C_{N_{(\phi'=22.5°)}} \sin(\phi') = f(\alpha', M^{\#})$$

$$CN = C'_{No} = C_N = f(\alpha', M^{\#})$$

$$CNP = \Delta C'_{N,\phi'} = C'_{N_{(\phi'=45°)}} - C'_{N_{(\phi'=0°)}} = f(\alpha', M^{\#})$$

$$C'_{N_{(\phi'=45°)}} = C_{N_{(\phi'=45°)}} \cos\phi' - C_{Y_{(\phi'=45°)}} \sin\phi'$$

$$CNDQ = C'_{N_{\delta q}} = \frac{\partial C_N}{\partial \delta q} \approx \frac{\Delta C_N}{\Delta \delta q} = f(M^{\#})$$

$$CLLAP = C_{l,\phi'_{\alpha}2} = \frac{\sum_{I=1}^{\#\alpha-1} \left[\frac{C_{l(\phi'=22.5°)} - C_{l(\phi'=0°)}}{\Delta \alpha^2} \right]}{(\#\alpha - 1)} = f(M^{\#})$$

$$C_l = C_{l,\phi'_{\alpha}2}\, \alpha'^2 \sin 4\phi'$$

$$CLLP = C_{l_p} = -2.15(Y_{CENT} / l_{REF}) C_{l_{\delta p}} = f(M^{\#})$$

$$CLLDP = C_{l_{\delta p}} = \left[\frac{C_{l_{(\delta p=5°,\alpha'=0°)}} - C_{l_{(\delta p=0°,\alpha'=0°)}}}{\delta p} \right] = f(M^{\#})$$

$$CLM = C'_m = C'_m = f(\alpha', M^{\#})$$

$$CLMP = \Delta C'_{m,\phi'} = C'_{m(\phi'=45°)} - C'_{m(\phi'=0°)} = f(\alpha', M^{\#})$$

$$C'_{m(\phi'=45°)} = C_{m(\phi'=45°)} \cos\phi' - C_{n(\phi'=45°)} \sin\phi'$$

$$C'_{m(\phi'=0°)} = C_{m(\phi'=0°)}$$

$$CLMQ = C'_{m_q} = (C_{mq_{(\alpha=0°)}} + C_{m\dot{\alpha}_{(\alpha=0°)}}) = f(M^{\#})$$

$$CLMDQ = C'_{m_{\delta q}} = \frac{\partial C_m}{\partial \delta q} \approx \frac{\Delta C_m}{\Delta \delta q} = f(M^{\#})$$

$$CLNP = \Delta C'_{n,\phi'} = C'_{n(\phi'=22.5°)} = f(\alpha', M^{\#})$$

$$C'_{n(\phi'=22.5°)} = C_{m(\phi'=22.5°)} \sin\phi' + C_{n(\phi'=22.5°)} \cos\phi'$$

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